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15 April 1967

Prepared by

R. B. Carpenter Jr.
Project Engineer
Mission Extension Studies

Approved by

B.J.Staub
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FOREWORD

This is Volume II of a four-volume report. Volumes III and IV will be available after 1 October 1967. This volume reflects the results of the first six-months study. It was conducted to determine the adequacy of basic Apollo equipment for manned interplanetary spacecraft and determine the most effective method of providing long-duration mission assurance.

The characteristics of several subsystems resulted in their elimination from further consideration. In these instances, subsystem configurations already proposed for similar missions were substituted. Subsystem evaluation for this phase of the study effort consisted of reliability/system effectiveness analyses augmented by RECOMP mission simulation, which resulted in quantitative estimates of mission reliability and definition of equipment areas where reliability is inadequate for manned interplanetary travel.

The study was conducted during the first half of CFY 1967 by the Systems Engineering Management Division of the Space and Information Systems Division of North American Aviation, Inc., under Research Authorizations (RA) 02195 - 15400 and 02195 - 15100. Documentation of the study is contracted under NAS2-4214, by the Mission Analysis Division, NASA/OART, Ames Research Center, Moffett Field, California.

The work was performed under the direction of Roy B. Carpenter, Jr., the program manager, Advanced Operations Analysis, Systems Engineering Management. Substantial contributions were made by H. L. Steverson, R. F. Wadsworth, L. K. Relyea, E. M. Murad, J. P. Goggins, J. A. Roebuck and I. Streimer.

NOTE: The "Background" and "Requirements for Maintenance" sections from Volume 1 have been included here to establish continuity of thought and to assure a proper understanding of the problem with which this study is concerned.

Because extended, manned, earth orbital and interplanetary flights pose a challenge to the nation's technological capability, reasonable assurance of mission success and crew safety is a basic prerequisite to these flights. It is self-evident that a need exists to create an integrated design and operational concept that will yield the desired safety assurance, and yet be feasible, economical, and acceptable to engineering and management. However, the no-failure-allowed approach to achieving mission assurance is unrealistic for the longer manned missions within the 1970-1980 period.

Since the ability to improve mission success has already been demonstrated, the demand has grown for a long-mission-duration manned spacecraft design that will make maximum use of both man and machine. Thus it is evident that the concept should include man as a maintenance expert, a trouble anticipating sensor, a backup operator, a backup computer, and perhaps other functions as yet undefined. In sum, such an approach is embodied in the availability concept.

This concept, originally applied to ground electronic systems, has been developed under a prior study for application to manned space missions, under a design and analysis technique deriving an optimum man-machine-mission relationship. It assures the required operational availability of spacecraft system functions while remaining within the constraints imposed by the crew, equipment, and mission commitments. The concept requires that the crew shall recognize degraded performance, isolate its cause, and perform the required maintenance action.

The maintenance aspect of the concept has been the subject of several former studies conducted at S&ID which indicate that man, given adequate preparation, can probably perform the required activities. Indications are that the maintenance workload would not exert a pronounced influence on crew utilization and it appears that the maintenance requirements can be identified with reasonable accuracy. Uncertainty relating to failure rates imposes a small weight penalty in increased spares, and its effect appears to be well below that of the uncertainty in propellant requirements.

The Apollo spacecraft or its subsystems were not specifically designed to facilitate in-flight maintenance. But they do represent what may be considered the 1970 technology. Apollo subsystems were used as the study baseline, as modified by the flyby mission requirement. Commonality of functions of primary, or crew-sensitive spacecraft systems* for diverse missions (planetary, lunar, or orbital) makes it reasonable to assume that the system components would be similar, if not interchangeable, for a given time period such as the 1970's or 1980's. In addition, the development problems associated with new designs for each new mission make such a philosophy unattractive because of both cost and risk.

The extended-mission Apollo represents the best contemporary source of detailed system design data upon which to base the proposed study. These data, in conjunction with the configurations established in the Mars/Venus Flyby Study (Ref. 1.1), provide the basis for the mission and systems for the planetary mission module. The planetary, lunar landing modules and earth recovery modules are expected to use the same system components, but the

*A system wherein loss of function would be deleterious to the crew.

need for maintainability is less crucial because of the relatively short duty cycles. By application of the same reasoning, the Apollo Extension Systems (Ref. 1.2) form a realistic base for the extended earth-orbital and lunar missions. (See Volume 1.)

The study was conducted in three phases. Phase I identifies the maintenance problem in terms of expected requirements and constraints; Phase II will include design sensitivity, and the required trade-off analysis; and Phase III will analyze the effects on spacecraft design and establish a mission system description as it applies to the reliability and crew safety problems.

S&ID has studied for several years reliability problems associated with extended manned space travel—a portion of the work being accomplished under NASA contracts (Refs. 1.1, 1.2, 1.3 and 1.4), and in addition, S&ID, through company-sponsored studies, has continued these efforts. These, in conjunction with the Apollo Program and its extension studies, have provided for the time period of interest a wealth of data on the reliability/crew safety aspects of manned spacecraft. (See Volume 1.)

As shown in Figure 1.1, the data indicates that, during the next decade, it will be impractical to attempt design of a spacecraft for maintenance-free operations for missions in excess of about 45 days. The practical mission limits for a non-maintainable design for a manned spacecraft probably vary between 30 to 45 days, depending on the mission profile and objectives. As missions are extended in duration and the abort profiles become more complex and time consuming, equipment failure becomes virtually certain. Further, a point is reached where adding redundancy no longer compensates for potential failures, but rather adds to the failure hazard. This technology limit is created by the need to include switching devices, performance monitors and voting circuits, as well as the connections to the system in the function reliability assessment. The practical limit seems to be between one and two additional components in simple redundancy. Maintenance beyond this point must be considered as a more reasonable alternative (Vol. 1).

This study is concerned with mission durations measured in years. The approximate mission reliability requirements in terms of mean time before failure (MTBF) without maintenance are:

1. Venus Flyby — 20 to 100 x 10^4 hours
2. Mars Flyby — 20 to 1000 x 10^4 hours
3. Mars Landing — 20 to 600 x 10^4 hours

The state-of-the-art capability has been shown (Ref. 1.1) to be as follows:

1. Without redundancy, approximately 0.1 to 1.0×10^4 hours

2. With optimum redundancy, approximately 1.0 to 5.0×10^4 hours.

If no failures are to be tolerated these estimates obviously fall far short of the expressed requirements, literally by orders of magnitude. Further, this same study indicated that, on the average, system MTBF can be improved by factors of between 5 and 10 over any decade. The effect of applying those systems to the longer space missions are demonstrated in Figure 1.1, where a state-of-art spacecraft is applied to the missions of interest without programmed maintenance.

Clearly, the longer missions must be prepared for failures. At this point, some study results have suggested that a possible alternative would include abort, spacecraft replacement, escape capsules, or rescue. But, S&ID studies have shown that, for the planetary and most lunar area missions, none of these will assure crew curvival (Refs, 1.1, and 1.5). These are reviewed later in this study report and shown to be ineffective. Conversely, provision for even the simplest of maintenance provides very startling results in terms of increases in mission success and crew survival (Vol. 1).

For example, the typical state-of-the-art spacecraft MTBF is estimated to be about 2800 hours. Now, assume that the mission duration is about 400 hours. Without any repair the probability of mission success (no failure) is only:

$$R = e - \frac{400}{2800}$$

$$= 0.870$$

By making provisions for just one repair, reliability, R, is increased to at least:

0.933 at the lower boundry

or

0.990 at the upper boundary,

depending on the assumed distribution, the method of calculation, and how the provision for the repair was implemented. Adding provisions for one more repair (in the critical system), or a total of two, raises the lower boundary estimate for mission reliability, R, to more than 0.99. These data indicate that providing for maintenance for the longer missions possesses a very attractive potential for increasing probability of mission success. Further, this is one case where the mathematics present a very conservative picture of the actual gains derived. This effect is dramatically shown in Figure 1-2 which presents an estimate of mission reliability as a function of mission duration and spares application. The lower curve, the baseline spacecraft, is representative of the latest AES reliability estimates derived from Apollo data. The curves above the base spacecraft represent the effects of adding one spare to the previous state for replacement of a critical component in the listed system. Note that a marked effect on mission reliability is achieved by adding only three spares. (See Volume 1.)

The effects of sparing on the probability of safe return are not as dramatic for many earth orbital missions because of the abort capability. However, for the extended lunar and planetary missions, the results of M&R actions are essentially the same as shown for mission success. This condition prevails because of the abort criteria applied to the Apollo missions and the very high initial probability of crew survival. But, as the missions are extended in distance away from the earth, the abort time delay exercises an increasingly more significant influence on the survival characteristics of a nonmaintainable manned spacecraft design.

From the foregoing, it is evident that a few spares and the associated M&R actions have a profound effect on both missions safety and success. The next most natural question is how many M&R actions are required to achieve a reasonable level of safety for the missions of interest? This is actually a major subject of the study. However, the number was estimated in Ref. 1.1. Using the prior estimated spacecraft MTBF, the number of M&R actions to be prepared for can be calculated as a function of the risk to be taken in meeting the need.

Since we are dealing with a statistical problem the answer will be statistical in nature; if the mission risk or reliability is to be 0.95, this means that a 5-percent risk of not having the required part for a failure is acceptable. In Figure 1.3, the number of M&R actions to be prepared for has been estimated for the state-fo-the-art spacecraft, as a function of mission duration and acceptable risk. With only a 5-percent risk, less than 85 M&R actions need be expected. Thus, this adds up to less than one activity per week of mission time, a very modest workload indeed. A basic intention of this study is to identify which specific components in a typical contemporary spacecraft will require attention and in what form. (Volume 1.)

In the past, maintenance has often been accomplished despite the design, rather than as a result of designing for maintenance. But now, crew safety, political, and cost aspects make it mandatory to take advantage of the potential reduction in risk inherent in the maintainable design. Since in-flight maintenance is dependent, to a large extent, on an amenable system configuration and hardware, full consideration must be given to the design for maintenance at the onset of the program. Thus, in-flight maintenance becomes an integral part of system design, and requires coordination with the concerned disciplines such as engineering, reliability, human factors, maintainability, logistics, and operations analysis. This systems approach has led to the availability concept detailed in Ref. 1.6 (Volume 1).

CONTENTS

ILLUSTRATIONS

TABLES

1. INTRODUCTION AND SUMMARY

SUMMARY OF RESULTS

This volume of the report contains the results of initial analyses conducted to determine the capability of subsystems to function on a Mars mission; the results of a computer simulation program conducted in conjunction with the analyses; and definition of problem areas requiring additional investigation. The basic mission timeline and equipment operating requirements are duplicated from Volume I and are used in the analyses contained in Section 2. Sections 3 through 9 contain discussions of individual subsystems. Potential problem areas and future study recommendations are in Section 10.

Results of analyses, as reflected in this volume, indicate that in general, the inherent reliability of the subsystems as now defined will not provide adequate assurance of mission success even with optimum redundancy. The combined electronics consisting of stabilization and control, communications, and guidance and navigation, result in an estimated reliability of 30 percent for the Mars Flyby Mission. This constitutes about a 68-percent contribution to total vehicle unreliability. Volumes III and IV reflect the results achieved by redesign, sparing, or in-flight maintenance. The major contributors to these low values are: 1. G&N Optics (G&N Subsystem), 2. SCS Attitude Electronics (SCS subsystem), 3. CDC (G&N, and 4. Star Tracker (Communications).

The overall inherent mission reliability for the Mars Flyby Mission Vehicle has been shown to be no better than 21 percent after optimizing redundancy and operational utilization. The approximate percent of unreliability contributed by the subsystems is provided in Table 1-1. This represents an increase in the probability of crew safety from about 0.03 to 0.21 through optimizing mission and systems design and yet without maintenance or repair.

STUDY OBJECTIVES

This study is being conducted in conjunction with and in support of the Manned Mars Flyby study NAS8-18025. The objectives covered in this volume are: (1) to determine the specific increases in system effectiveness/reliability of Apollo type equipments required for lunar and interplanetary missions of up to 700 days in duration; and (2) to define methods for achieving the required increases. The tasks covered in this volume are: (1) to define mission requirements and constraints, including mission timeline timelines, mission functional tasks, equipment operating requirements, down-time constraints, and abort criteria; (2) to conduct a contingency analysis in which alternate operating modes are considered; and (3) to conduct mission modeling and analysis by means of a RECOMP II program developed to provide quantitative reliability predictions and define areas/items contributing most to unreliability.

Table 1-1. Overall Inherent Mission Reliability

Subsystem	Percent Reliability	Percent Contribution To Vehicle Unreliability
Electronics	30.0	68.0
ECS/LSS	85.0	14.5
SPS	90.0	10.0
Spin Engine	95.0	5.0
Remainder	97.0	2.5
Vehicle Reliability	21.0	100.0

Approach: Since the baseline subsystems reflected Apollo equipment, and since a certain form of logic representing these subsystems was required for computer simulation, reliability block diagrams representing Apollo Block II configurations were developed. The Apollo environmental control/life support and the electrical power subsystems obviously are inadequate for the longer missions; therefore configurations proposed by the design groups supporting the study effort were used. Apollo subsystem logic differs depending on whether the criterion employed applies to mission success or crew safety. For example, two units, either of which is essential to system operation, would be represented in a series arrangement for mission success, since failure of either one would necessitate an abort. Crew safety logic would represent the units in parallel, since, in the event of one failure, the other unit would be available for system operation. In the case of a Mars mission, this criterion would apply up to Mars trajectory injection, since the mission could be aborted up to that point. After injection into a Mars trajectory, however, the ΔV requirements for abort exceed the capability of the on-board propulsion units, and abort, in effect, requires a Mars flyby and return. Crew safety logic was therefore used in evaluating the inherent reliability of subsystems after injection into the Mars trajectory. In the event of equipment failure, reliance must be placed on the redundant units, alternate operating modes, repair capability, or a combination of all three.

It is recognized that certain failures may cause curtailment of scientific experiments and impairment of mission objectives; however, the probability of such failures and the effect of these failures on the planned scientific workload is a subject for future study.

As an ancillary task to the subsystem reliability evaluation, each mission function was examined from the aspect of criticality and alternate equipment use. Mission success, defined as the accomplishment of all objectives with safety for the crew was considered in determining the dependence on the individual functions. A summary of this evaluation is contained in Appendix A.

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2. MISSION TIMELINES

Tables 2-1, 2-2, and 2-3 define the details of the mission timelines used as a reference for subsystem analyses. The total mission consists of 16,801.2 hours. The first three tables contain subsystem operating/ cyclic requirements for the twelve basic mission phases. The two longest phases, consisting of transit to Mars and return, have been further subdivided into second-level phases. They are shown on the fourth and fifth tables.

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Table 2-1. Top Level, System Function Duty Cycle Estimates (Part I)

Block Number	1.0	3.1 to 3.4	3.5	3.6	3.7	3.8	3.9	3.10	3.11	3.12	3.13	3.14 3.15	
Mission Phase	Prelaunch	Ascent to Orbit Injection	Earth Orbit	Trans-Planet Injection	Trans-Planet Coast	Planet Approach	Planet Encounter	Trans-Earth	Earth Approach	Earth Retro	Earth Entry	Recovery	Total System Usage
Phase Duration	96.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	72.0	16,801.2
Communication													
HF Transceiver	0.5	- 0 -	1.0	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	12	13.05
VHF/AM Transceiver	0.5	12.0	24.0	0.70	33.6	2.0	460	12.7	8.0	0.10	0.36	-	553.96
VHF Beacon	0.5	- 0 -	1.0	0.70	8.0	- 0 -	- 0 -	- 0 -	8.0	0.10	0.36	72	90.66
S-Band Tran and Amplifier	0.5	- 0 -	1.0	- 0 -	336	10.0	230	1270	10.0	- 0 -	-	-	1,847.5
Unified "S" Band Rec.	0.5	- 0 -	1.0	- 0 -	3360	20.0	460	12,700	- 0 -	- 0 -	-	-	16,551.5
C-Band Transponder	0.5	24	48.0	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	20.0	0.10	0.36	-	82.96
Antenna													
DSIF High Gain	- 0 -	- 0 -	0.30	- 0 -	3360	20.0	460	12,700	- 0 -	- 0 -	-	-	16,551.5
UHF Multiplexer	1.0	12	24	0.70	- 0 -	10.0	460	48.0	8.0	0.10	0.36	-	564.16
VHF Omni	1.0	24.0	24	0.70	- 0 -	10.0	460	48.0	8.0	0.10	0.36	-	576.16
VHF Switch (Cycles)	10 cy	- 0 -	4 cy	- 0 -	- 0 -	30 cy	1380 cy	144 cy	4 cy	- 0 -	- 0 -	-	1,572.0cy
Data Handling													
Audio Center	5.0	24.0	48.0	0.70	3360	20.0	460.0	12,700	20.0	0.10	0.36	12.0	16,710.2
Premod. Processor	3.0	12.0	24.0	0.70	1120	20.0	460	4,250	10.0	- 0 -		-	5,899.70
Data Storage	3.0	- 0 -	24.0	- 0 -	1680	20	460	6,350	8.0	- 0 -	-	-	8,545.0
PCM Telemetry	3.0	24.0	24.0	0.70	- 0 -	10	460	48.0	10.0	-	-	-	565.70
TV Camera	1.0	- 0 -	12.0	- 0 -	8.0	- 0 -	24	8.0	1.0	-	-	-	54.0
Central Timer	5.0	24.0	48.0	0.70	3360	20	460	12,700	20.0	0.10	0.36	-	17,560.46
Up Data Link	3.0	24.0	24.0	0.70	- 0 -	10	460	8.0	10.0	- 0 -	0.36	-	530.16
Teleprinter	1.0	- 0 -	0.5	- 0 -	33.6	- 0 -	1.0	127	- 0 -	- 0 -	-	-	163.1
Signal Conditioner	3.0	24.0	24.0	0.70	1120	20	460	4,250	10.0	0.10	0.10	-	5,911.90
Guidance and Navigation													
Guidance Computer	4.0	24	12	0.70	168	18.0	400	168	18.0	0.10	0.36	-	813.16
Inertial Measure	4.0	24	24	0.70	168	18.0	400	168	18.0	0.10	0.36	-	825.16
Optical Alignment	0.1	2	2	- 0 -	1.0	0.30	1.0	1.0	0.30	- 0 -	- 0 -	-	7.70
Displays and Control	4.0	24	24	0.70	168	18.0	400	168	18.0	0.10	0.36	-	825.16

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 2-1. Top Level, System Function Duty Cycle Estimates (Part I) (Cont)

Block Number	1.0	3.1 to 3.4	3.5	3.6	3.7	3.8	3.9	3.10	3.11	3.12	3.13	3.14 3.15	
Mission Phase	Prelaunch	Ascent to Orbit Injection	Earth Orbit	Trans-Planet Injection	Trans-Planet Coast	Planet Approach	Planet Encounter	Trans-Earth	Earth Approach	Earth Retro	Earth Entry	Recovery	Total System Usage
Phase Duration	96.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	72.0	16,801.2
Spin - Despin Control													
Electronics	1.5	- 0 -	0.30	- 0 -	3190	1.2	- 0 -	12,532	1.2	-	-	-	15,727.2
Propellant Storage	18	24.0	0.70	0.70	3190	20.0	460	12,532	20.0	-	-	-	16,312.7
Propellant Management	1.5	- 0 -	0.30	- 0 -	3190	1.2	- 0 -	12,532	1.2	-	-	-	
Engine - Cycles (Sp)	1 cy	- 0 -	1 cy	- 0 -	7 cy	1 cy	- 0 -	7 cy	1 cy	-	-	-	16 cycles
- Burn		- 0 -		- 0 -	4090 sec	1250 sec	- 0 -	3,615 sec	1000 sec	-	-	-	9,995 sec
Wabble Damper	1.5	- 0 -	0.30	- 0 -	3350	1.2	- 0 -	12,532	1.2	-	-	-	15,886.2
Precess Engines (cy)	1 cy	- 0 -	1 cy	- 0 -	8 cy	- 0 -	- 0 -	10 cy	- 0 -	-	-	-	20 cycles
- Burn	2 sec	- 0 -	2 sec	- 0 -	175 sec	- 0 -	- 0 -	200 sec	- 0 -	-	-	-	379 sec
Mechanical Systems													
S/C Structure (MM)	- 0 -	- 0 -	46.0	0.70	3360	20.0	460	12,700	20.0	0.10	-	-	16,606.8
Centrifuge	0.2	- 0 -	- 0 -	- 0 -	560	- 0 -	- 0 -	2,120	- 0 -	-	-	-	2,680.2
Ext. - Ret. Cont.	0.5	- 0 -	0.30	- 0 -	3350	1.2	- 0 -	12,690	1.2	-	-	-	16,043.2
Ext. Ret. Mech	- 0 -	- 0 -	0.30	- 0 -	3350	1.2	- 0 -	12,690	1.2	-	-	-	16,042.7
Docking Mech	- 0 -	- 0 -	- 0 -	- 0 -	10.0	18.8	460	10.0	20.0	0.10	-	-	518.9
Reentry Module	- 0 -	24.0	2.0	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	- 0 -	0.10	0.36	72	98.4
Storm Cellar	- 0 -	- 0 -	- 0 -	- 0 -	(3.4)	- 0 -	- 0 -	(127)	-	-	-	-	(130 Max)
Performance Monitor	3.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	-	-	-	16,635.7
Diagnostic Systems	0.5	- 0 -	2.0	- 0 -	80	1.0	10.2	302	1.0	-	-	-	396.7
S/C Propulsion													
Propellant Storage	18.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.1	-	-	16,602.8
Propellant Management	1.5	2.0	2.0	- 0 -	- 0 -	2.6	- 0 -	2.0	- 0 -	0.1	-	-	9.6
Engine - Cycles	- 0 -	- 0 -	1 cy	- 0 -	1 cy	2 cy	- 0 -	2 cy	- 0 -	1 cy	-	-	7 cycles(Max)
- Burn	- 0 -	- 0 -	0.5 sec	- 0 -	25 sec	50 sec	- 0 -	50 sec	- 0 -	25 sec	-	-	155 sec
Electrical Power													
Isotope Source	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
Energy Conversion	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
Processing and Distribution	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
Storage - Cycles	1 cy	- 0 -	5 cy	- 0 -	336 cy	10 cy	230 cy	1,270 cy	2 cy	- 0 -	- 0 -	- 0 -	1,854 cy
- Duration	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	72.0	16,710.2

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 2-1. Top Level, System Function Duty Cycle Estimates (Part I (Cont)

Block Number	1.0	3.1 to 3.4	3.5	3.6	3.7	3.8	3.9	3.10	3.11	3.12	3.13	3.14 3.15	
Mission Phase	Prelaunch	Ascent to Orbit Injection	Earth Orbit	Trans-Planet Injection	Trans-Planet Coast	Planet Approach	Planet Encounter	Trans-Earth	Earth Approach	Earth Retro	Earth Entry	Recovery	Total System Usage
Phase Duration	96.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	72.0	16,801.2
Stability Control													
Electronics	3.0	24.0	48.0	0.70	168	18.8	460	168	18.8	0.10	-	-	909.4
Fuel Storage	18	24.0	48.0	0.70	3360	20.0	460	12,700	18.8	0.10	-	-	16,649.6
Fuel Feed (cycles)	1 cy	100 cy	150 cy	2 cy	602 cy	400 cy	8,000 cy	602 cy	200 cy	2 cy	-	-	10,059 cy
Reaction Eng. (cycles)	1 cy	100 cy	150 cy	2 cy	602 cy	400 cy	8,000 cy	602 cy	200 cy	2 cy	-	-	10,059 cy
Manual Control	0.2	- 0 -	0.5	- 0 -	- 0 -	2.0	46.0	- 0 -	2.0	0.10	-	-	50.8
Life Support (L.S)													
●Cabin Temperature	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
O_2 Supply and Control	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
N_2 Supply and Control	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
Waste Management	0	24.0	48.0	- 0 -	3360	20.0	460	12,700	20.0	- 0 -	-	-	16,632.0
●CO_2 Removal	0.5	24.0	48.0	- 0 -	3360	20.0	460	12,700	20.0	- 0 -	-	-	16,632.0
Trace Cont. Removal	0.5	24.0	48.0	- 0 -	3360	20.0	460	12,700	20.0	- 0 -	-	-	16,632.0
Water Management	0.2	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	- 0 -	-	-	16,632.0
Personnel Hygiene	0.2	- 0 -	4.0	- 0 -	786	2.0	19	2,180	4.0	- 0 -	-	-	16,633.0
Food Management	0.2	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	- 0 -	-	-	16,638.2
Personal L.S. (Portable)	0	- 0 -	4.0	- 0 -	- 0 -	4.0	- 0 -	2.0	2.0	- 0 -	-	-	12.0 (Max)
Environment Control													
●Coolant (Equip) Temp.	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2
Atmospheric Pressure	3.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,636.2
●Humidity Cont.	3.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,636.2
Space Suits	3.0	24.0	24.0	- 0 -	- 0 -	4.0	- 0 -	2.0	2.0	0.10	0.36	- 0 -	59.6 (Max)
●Isotope Cooling	5.0	24.0	48.0	0.70	3360	20.0	460	12,700	20.0	0.10	0.36	-	16,638.2

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 2-2. Second Level Duty Cycle Estimates, 3.7 Transplanetary Phase

Block Number	3.7.1 - 6	3.7.7	3.7.8 - 10	3.7.11	3.7.12 - 14	3.7.15	3.7.16 - 18	
Mission Phase	Zero G Coast	Spin-up Spacecraft	Spin Coast	De-Spin Spacecraft	Mid-Course Correction	Spin-up Spacecraft	Spin Coast	Total System Usage
Sub Phase Duration	168	1.2	1515.7	1.2	168	1.2	1515.7	3360
Stability Control								
Electronics	168	- 0 -	- 0 -	- 0 -	168	- 0 -	- 0 -	336
Fuel Storage	168	1.2	1515.2	1.2	168	1.2	1515.7	3360
Reaction Eng. Cyc.	301 cy	- 0 -	- 0 -	- 0 -	301 cy	- 0 -	- 0 -	602 cy
Fuel Feed - Cyc.	301 cy	- 0 -	- 0 -	- 0 -	301 cy	- 0 -	- 0 -	602 cy
S/C Propulsion								
Propellant Storage	168	1.2	1515.2	1.2	168	1.2	1515.2	3360
Propellant Management	20*	- 0 -	- 0 -	- 0 -	20*	- 0 -	- 0 -	40
Engine - cycles	2*	- 0 -	- 0 -	- 0 -	1*	- 0 -	- 0 -	3 cy
- burn	25sec*	- 0 -	- 0 -	- 0 -	25sec*	- 0 -	- 0 -	50 sec
Spin - Despin Control								
Electronics	- 0 -	1.2	1515.2	1.2	- 0 -	1.2	1515.2	2034.
Propellant Storage	168	1.2	1515.2	1.2	168	1.2	1515.2	3360
Propellant Management	- 0 -	1.2	1515.2	1.2	- 0 -	1.2	1515.2	2034
Spin Eng. - Cycles	- 0 -	1 cy	2 cy	1 cy	- 0 -	1 cy	2 cy	7 cy
- burn	- 0 -	1400 sec	20 sec	1350 sec	- 0 -	1300 sec	20 sec	4090 sec
Precess Eng. - cycle	- 0 -	- 0 -	4 cy	- 0 -	- 0 -	- 0 -	4 cy	8 cy
- burn	- 0 -	- 0 -	75 sec	- 0 -	- 0 -	- 0 -	75 sec	150 sec
Wabble Damper	- 0 -	1.2	1515.2	1.2	- 0 -	1.2	1515.2	2034
Guidance and Navigation								
Guidance Computer	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168
Inertial Measure	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168
Display & Cont.	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168

*Worst Case Assumed

Table 2-3. Second Level Duty Cycle Estimates, 3.10 Transearth Phase

Block Number	3.10.1 - 6	3.10.7	3.10.8 - 11	3.10.12	3.10.13 - 15	3.10.16	3.10.17 - 19	
Mission Phase	Zero G Coast	Spin-up Spacecraft	Spin Coast	De-Spin Spacecraft	Mid-Course Correction	Spin-up Spacecraft	Spin Coast	Total System Usage
Sub Phase Duration	168	1.2	6,185.2	1.2	168	1.2	6,185.2	12,700
Stability Control								
Electronics	168	- 0 -	- 0 -	- 0 -	168	- 0 -	- 0 -	336
Fuel Storage	168	1.2	6,185.2	1.2	168	1.2	6,185.2	12,700
Reaction Eng. - cyc	301 cy	- 0 -	- 0 -	- 0 -	301 cy	- 0 -	- 0 -	602 cy
Fuel Feed - cyc	301 cy	- 0 -	- 0 -	- 0 -	301 cy	- 0 -	- 0 -	602 cy
S/C Propulsion								
Propellant Storage	168	1.2	6,185.2	1.2	168	1.2	6,185.2	12,700
Propellant Management	2.0*	- 0 -	- 0 -	- 0 -	20*	- 0 -	- 0 -	22
Engine - cycles	2 cy*	- 0 -	- 0 -	- 0 -	1 cy*	- 0 -	- 0 -	3 cy
- burn	25 sec*	- 0 -	- 0 -	- 0 -	25 sec*	- 0 -	- 0 -	50sec
Spin - De-Spin Control								
Electronics	- 0 -	1.2	6,185.2	1.2	- 0 -	1.2	6,185.2	12,374
Propellant Storage	168	1.2	6,185.2	1.2	168	1.2	6,185.2	12,700
Propellant Management	- 0 -	1.2	6,185.2	1.2	- 0 -	1.2	6,185.2	12,374
Spin Eng. - cycles	- 0 -	1 cy	2 cy	1 cy	- 0 -	1 cy	2 cy	7 cy
- burn	- 0 -	1200 sec	85 sec	1150 sec	- 0 -	1100 sec	80 sec	3615sec
Precess Eng. - cycles	- 0 -	- 0 -	5 cy	- 0 -	- 0 -	- 0 -	5 cy	10 cy
- burn	- 0 -	- 0 -	100 sec	- 0 -	- 0 -	- 0 -	100 sec	200sec
Wabble Damper	- 0 -	1.2	6,185.2	1.2	- 0 -	1.2	6,185.2	12,374
Guidance and Navigation								
Guidance Computer	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168
Inertial Measure	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168
Displays and Cont.	84	- 0 -	- 0 -	- 0 -	84	- 0 -	- 0 -	168

*Worst Case Assumed

3. SPIN ENGINE SUBSYSTEM

FUNCTION

A mission requirement providing for biological stability is that the entire spacecraft must be spinning during the long transplanetary coast periods. Prolonged exposure to zero g environment causes undesirable chemical changes within man, with the effect of progressive weakness. Prevention of this debility can be accomplished by imparting rotation to the vehicle in the extended configuration and thus creating an artificial gravity acceleration.

Energy for spin-up and de-spin is provided by a spin engine subsystem using directed products of combustion to increase and decrease rotation. Figure 3-1 contains a subsystem schematic. A list of major components is provided. The current method requires spinning of the spacecraft in a mated condition, then extending and increasing speed of rotation until astronaut comfort is achieved. De-spin would be the reverse procedure and is absolutely required to accomplish midcourse correction, planetary encounter operations, and reentry corrider establishment.

DEFINITION OF DUTY CYCLES

Thrust is required for one burn cycle at each extension and retraction. The direction is predetermined: no gimballing is required. The time period depends on the spacecraft and engine sizes and can vary over a wide range. Valve timing is not critical because the engine(s) can be started at manual command, left operating for long periods, and shutdown when acceptable spin is achieved within wide tolerances.

Although thrusting occurs only occasionally during the flight, most of the system is required to prevent propellant and pressurization fluid loss at all times. The system is entirely filled with propellants from the initial extension immediately after transplanetary injection. Leakage at any point would degrade subsequent operation in the ability to cope with the requirements of emergency use. It is expected that redundant sets of propellant containers will be on board; so that the capability of mission completion will be retained despite an individual loss.

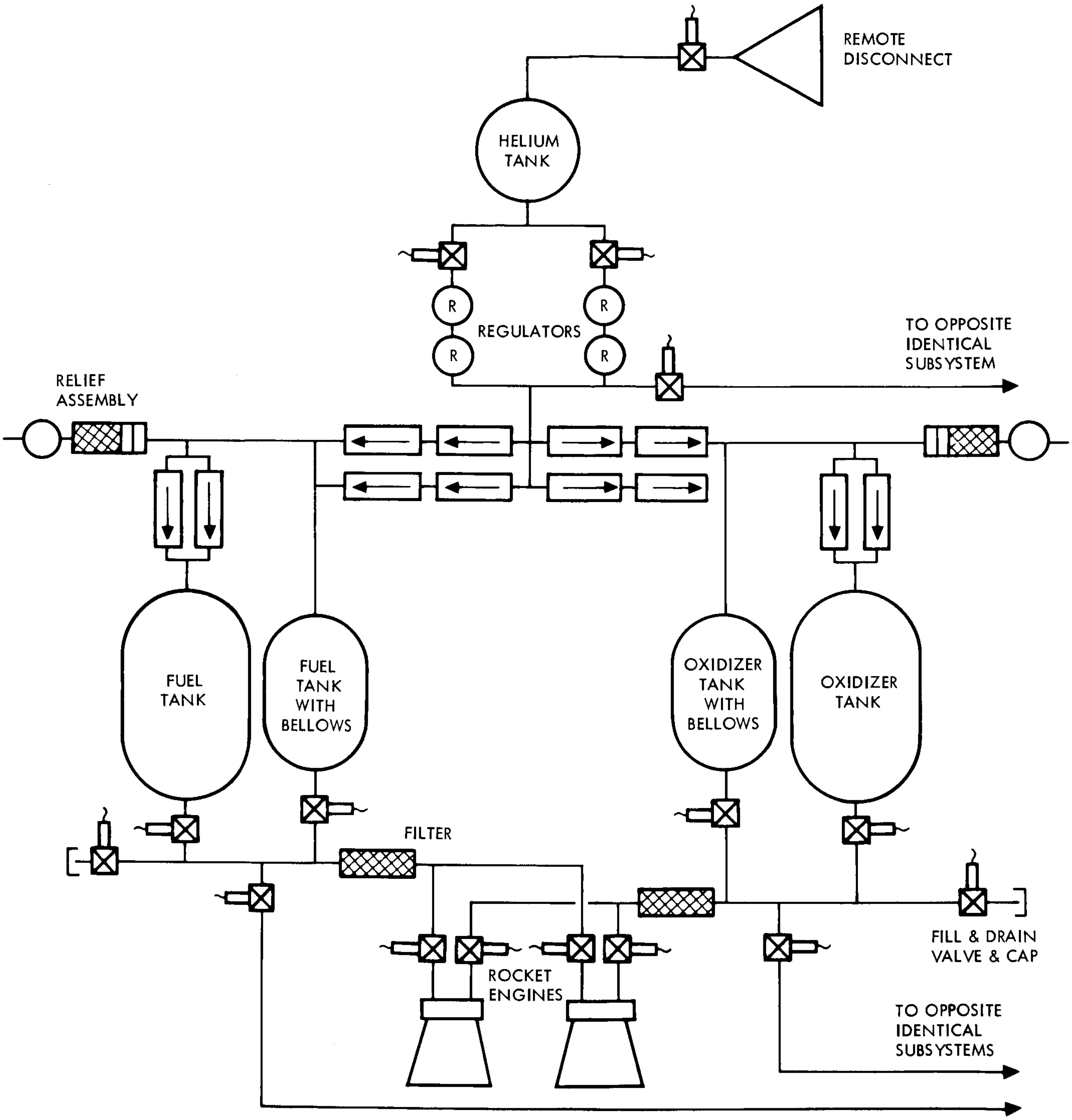

Figure 3-1. Spin Engine Subsystem Schematic

The following duty cycle has been taken from the mission timeline previously presented:

Phase 3.1 through 3.4	Propellant storage throughout 24 hours
Phase 3.5	Propellant storage throughout 48 hours
Phase 3.6	Propellant storage through 0.7 hours
Phase 3.7	Propellant storage throughout 3360 hours
	For 7 engine burn cycles, there will be 4 extensions and 3 retractions
Phase 3.8	Propellant storage throughout 20.0 hours
	For one engine burn cycle, there will be one retraction
	Electronics operating only during spin for 12.0 hours
Phase 3.9	Propellant storage throughout 460 hours
Phase 3.10	Propellant storage throughout 12,700 hours
	For 7 engine burn cycles, there will be 4 extensions and 3 retractions
	Electronics operating only during spin for 12,532 hours
Phase 3.11	Propellant storage throughout 20 hours
	For one engine burn cycle, there will be one retraction
	Electronics operating only during spin for 1.2 hours

All components will be subjected to vacuum environment throughout the flight, the engines external and exposed to meteorite impingement and space radiation.

Reaction Control Subsystem Component List For
Artificial Gravity Spin,
Including Control Electronics

Two Apollo "Quads" with propellant interconnection

Two engines per "Quad"

Spin Propulsion

2 Helium Tanks
2 Fill Valves and Disconnects
6 Helium Solenoid Valves
8 Regulators
4 Burst Discs and Filters
4 Relief Valves
4 Fuel Tanks
4 Oxidizer Tanks
12 Propellant Solenoid Valves
4 Main Filters
4 Fill Valves and Caps
4 Rocket Engines
12 Check Valves

IDENTIFICATION OF CRITICAL FUNCTIONS AND COMPONENTS

Even though the functions of providing either spin-up and de-spin forces or exercises are critical to crew health, no single component failure is expected to cause crew loss. Sufficient internal redundancy and the placement of propellant supplies on opposite sides of the spacecraft will prevent catastrophic loss from a single failure. The pressurizing gas, fuel, and oxidizer storage reach adequate reliability levels through redundancy; however, the exposure of the rocket engines to meteoroid bombardment is unnecessary. Special development of components for long-term propellant compatibility, addition of an alternate pressurization method, and protection of engines during long inactive periods are recommended for reliability improvement.

Discussion of Redundancy Available

The candidate subsystem, evaluated for effects of each component failure on the mission, is concisely presented in the reliability logic diagram (Figure 3-2). Either helium supply can pressurize all propellants through the interconnecting piping and isolation solenoids. Within each helium supply system, the helium fill valve must be operated remotely because of safety regulations. It is redundant with the quick disconnect to seal pressure within the tank during flight. The regulators and solenoid valves are redundant for the open and closed failure modes. Some protection against a small external leak may be gained by closing the solenoid valves between operations.

Since the fuel supply system is identical in logic with the oxidizer supply system, the problems and redundancies are much the same. Check valves are redundant between the helium supply and each propellant, so that hypergolic vapors cannot mix. The secondary check valves to the non-bellows tank are shown redundant in the flow direction only because vapor return is not serious. Liquid entrapped in the gas side of the partially empty bellows tank can be returned if the bellows is designed with a residual spring force. The isolating propellant solenoids would have to be kept open somewhat longer than minimum to allow transfer time. No relief valve operation is anticipated at normal operating pressures. Actuation is predicated on rupture of the burst disc which is designed to break above these pressures. Some positive means for prevention of propellant leakage through the fill valve is shown by the standby cap. Since it is expected that tank pressure will be low during filling, a cap manually placed is sufficient.

Discussion of Critical, Low Reliability Assemblies

For this report, criticality will continue to be evaluated at three levels:

1. Single failure mode resulting in possible crew loss.

2. Single failure mode resulting in possible mission loss.

3. Other considerations.

Some conditions may be separated into terms of crew safety and mission success, even on a long interplanetary mission. For this spin engine subsystem, a major mission success category occurs in earth orbit prior to transplanetary injection in the form of any nonrepairable failure which will cause mission abort, or at least a change to another vehicle. Another mission success condition is the extension process. If extension is not accomplished, a severe series of exercises will be required to retain sufficient bodily strength to withstand re-entry. The capacity of the exercises to produce the desired strength has not yet been proven, since the longest flight to date is 13 days, but exercise and drugs are presumed to be

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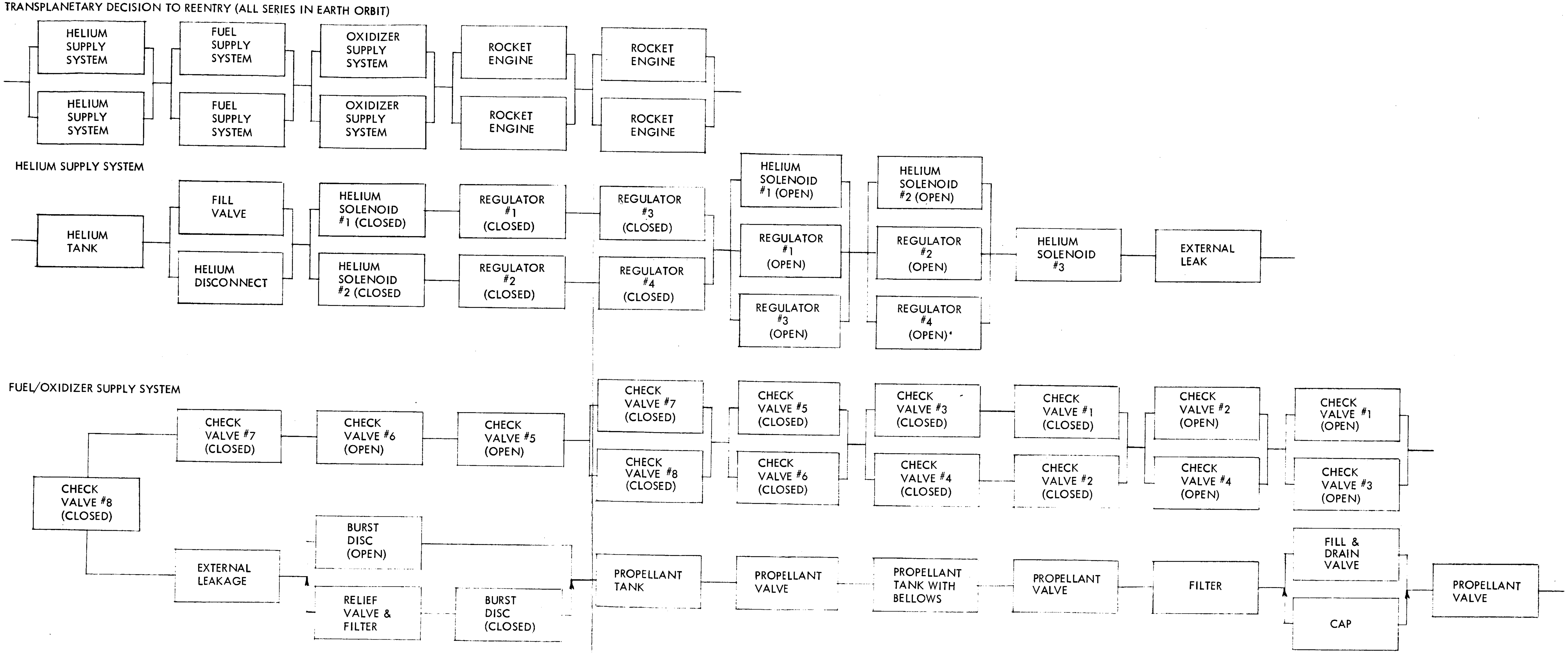

Figure 3-2. Spin Engine Subsystem Reliability Logic

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3

proven by the time of interplanetary flight. On the other hand, retraction is critical to crew members because of their inability to make mid-course corrections and to acquire the re-entry corridor.

No single component nor assembly is critical or has low reliability. Redundancy is provided throughout the system. However, return-trip meteoroid bombardment in the severe flux region may cause some difficulty. Even though knowledge of this environment is very imprecise, the fact of some risk is established. Since this spin engine system is only used during actual spin-up and de-spin of less than one hour each, continued exposure throughout the several thousand of hours coast is unnecessary and adds to the risk. A protective covering could be extended to cover these engines during coast.

Although redundant, the helium pressurizing gas is extremely difficult to retain during the long mission, not only because of changes in the seal but because of molecular penetration of thin walled materials. An alternate system of pressurization would be required if tests continue to show excessive permeation. Several methods (Figures 3-3, -4, and -5) based on gas generation from combustion products are being considered at the present time including:

1. Separate combustion chambers for fuel-rich and oxidizer-rich gas generation. (Pulsing-using check valves or continuous-using turbine and pumps.)

2. One fuel-rich gas generator and an oxidizer heat exchanger (pulsing, using check valves; or continuous, using turbine and pumps.

3. Direct injection of propellant into opposite hypergol tanks from a higher pressure source.

4. A central emergency pressure supply for all propulsion subsystems, using any one of a number of hypergolic propellant gas generators.

SPECIAL CONSIDERATIONS

The spin engine subsystem must be operated at the same time as the extension mechanism and the electronics necessary to determine the spin attitude. No interface problems are anticipated, since timing is not critical; i. e., there are long operating times with wide cut-off tolerances.

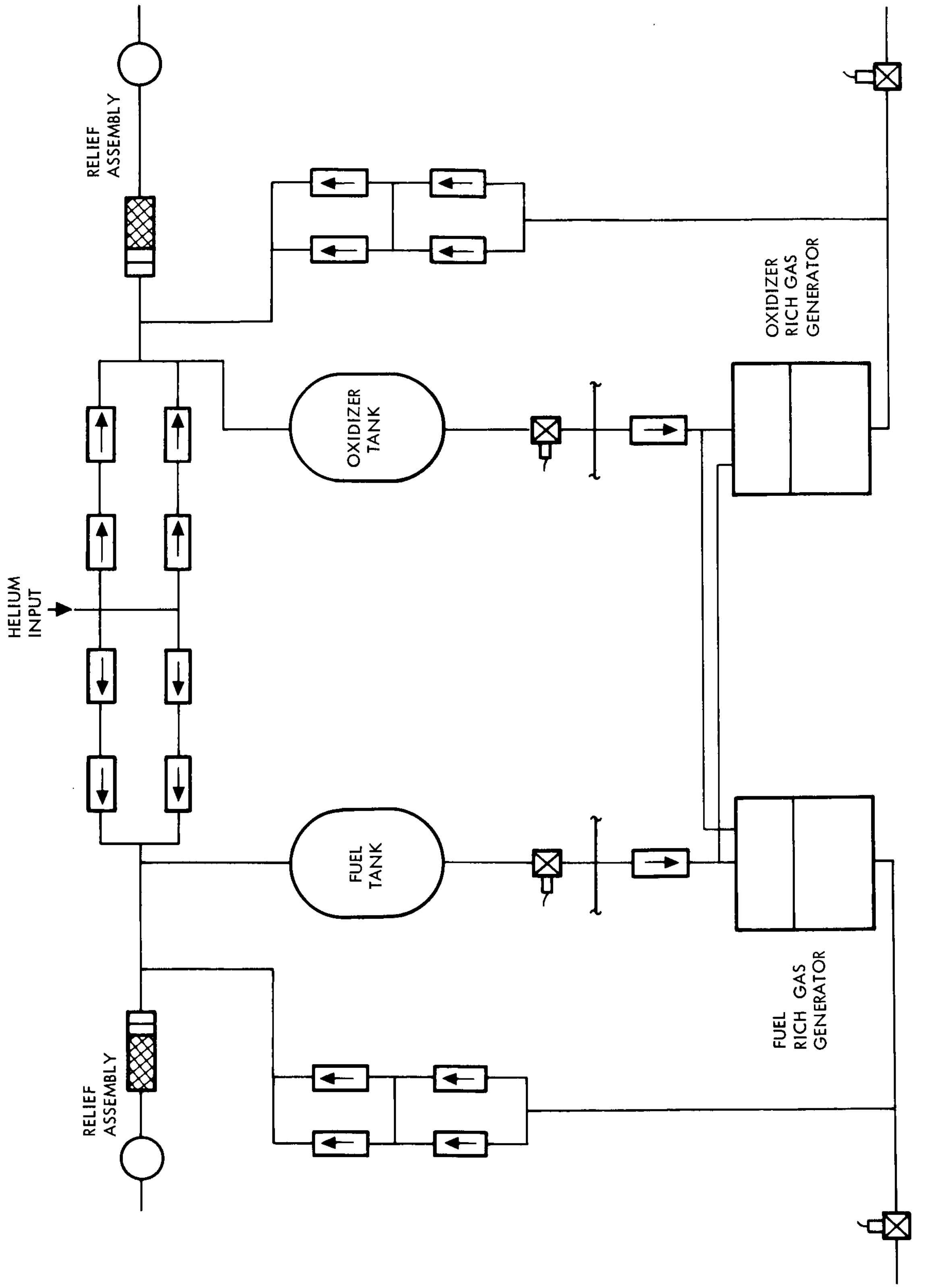

Figure 3-3. Gas Generation Concept with Separate Combustion Chambers

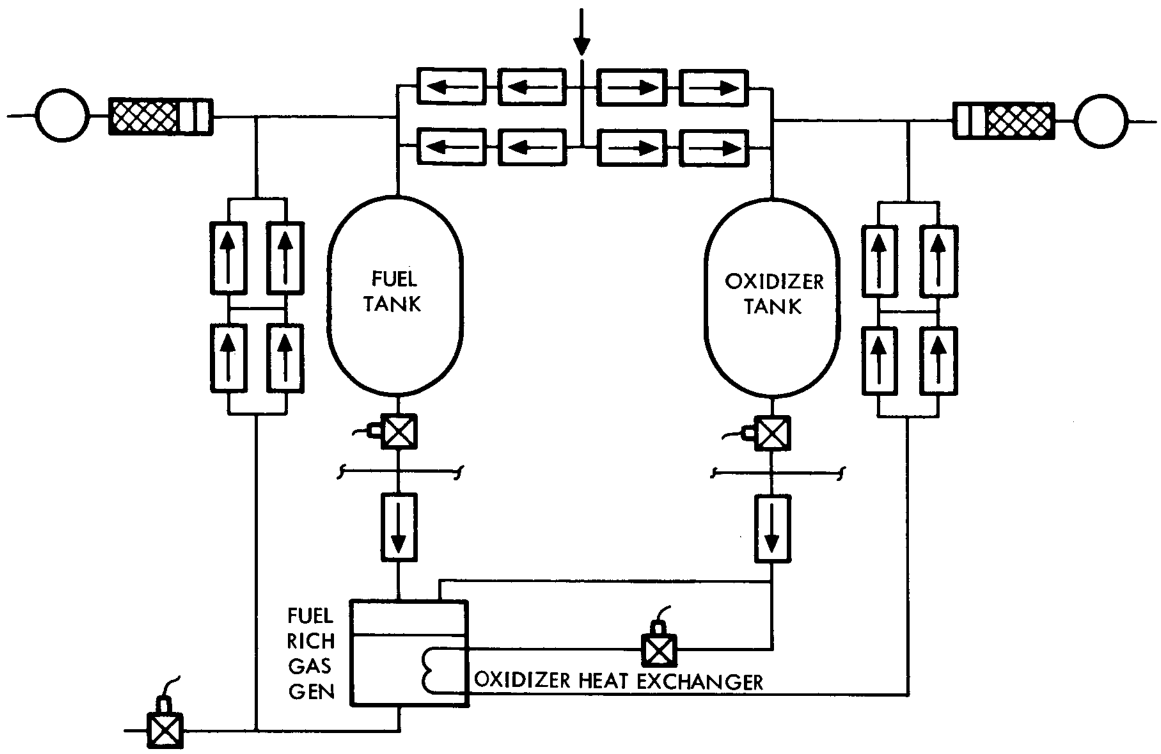

Figure 3-4. Gas Generation and Heat Exchanger

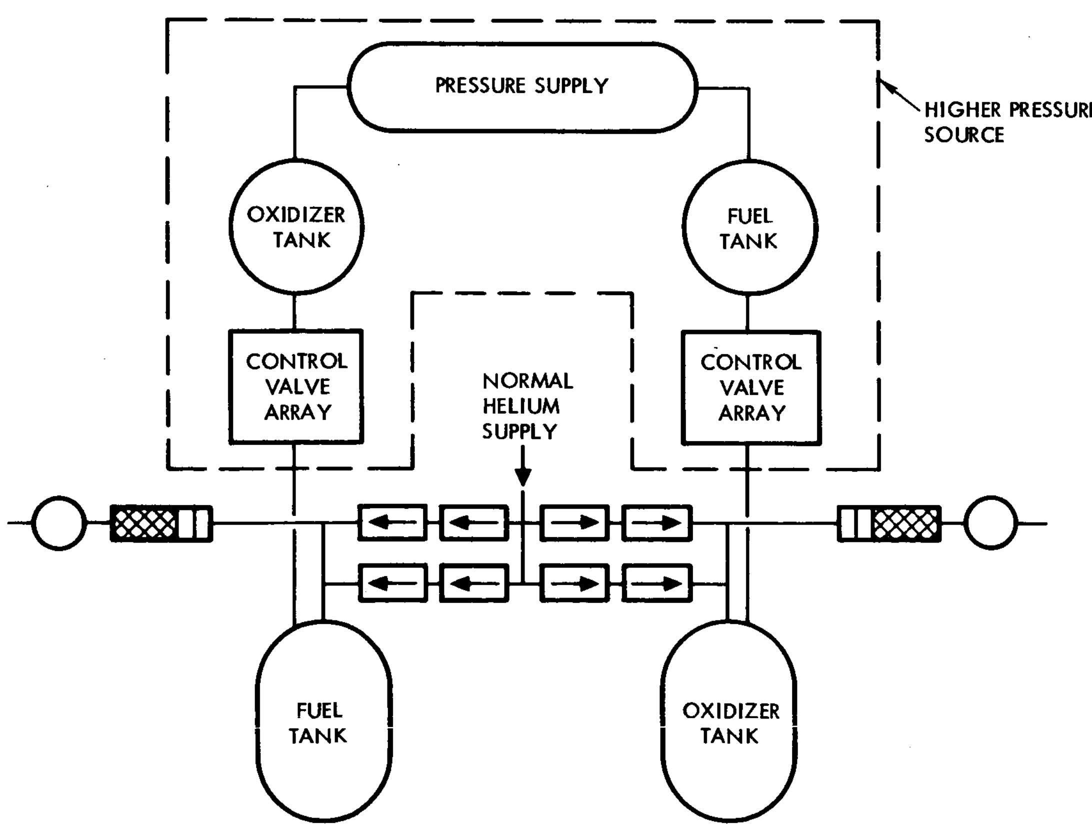

Figure 3-5. Separate Propellant Injection Concept

Development implications of the recommended alternate configurations are:

1. Positive expulsion device bias to return to full tank after each operation will not require extensive development.

2. Engine covering for meteoroid protection can be designed on a current technology basis but may be heavy. Development of meteroid bumpers should continue with design freeze only when production is necessary. Additional data should be obtained of the meteoroid flux to be encountered; so that a more optimum weight can be determined.

3. Continuing study of human tolerances to space environment is required to determine the criticality of the artificial gravity mode. The present uncertainty of long duration degradation during zero g must be dispelled by actual test in earth orbit.

4. Helium storage and alternate pressurization means should continue to be developed. The current benefits of helium are sufficient to stimulate containment research; however, an emergency system should be available for long, independent missions.

5. Positive expulsion is being accomplished by a heavy stainless steel bellows. Continuing development of an impervious flexible bladder will decrease the required equipment weight and increase the weight available for experiments.

Extensive earth-based facilities are not required; however, long duration, zero g environments in earth orbit are achieved only by a special spacecraft, and determination of meteoroid fluxes throughout the flight path requires special space probes. The earth orbit experiments must be manned, whereas the space probes can be unmanned.

RESULTS OF SUBSYSTEM ANALYSIS

The spin engine subsystem is currently required because of uncertainty over human tolerance to zero-g space conditions. If alternate tolerance methods become available, the entire artificial gravity system may be modified.

Spin engines are presently the most feasible way to achieve the thrust for spin-up and de-spin. Reasonably high performance within acceptable operating times is available from current engines; and by suitable redundancy, the reliability can be made acceptable even for the long durations.

The current reliability logic is too complicated for simulation by the Recomp computer; however, a gross estimate indicates a subsystem reliability near 95 percent. The problems are generally caused by components characteristics rather than by considerations of the overall configuration. For a higher required goal, small improvement in both areas will result in large subsystem reliability improvements.

5. REACTION CONTROL SUBSYSTEM

FUNCTION

Spacecraft attitudes during zero g must be controllable automatically and on command for a variety of functions, including: initial orientation for an impulse, roll control during an impulse, fine orientation for astronomical observations, and docking maneuvers for earth orbit assembly and returning planetary probes. Normal attitude, automatic hold will free the astronauts for attention to other operations without excessive disorientation, with the fine control reserved for special sightings. Direct astronaut control provides the flexibility necessary for unprogrammed, terminal operations and emergency maneuvers.

The reaction control subsystem provides the thrust necessary to move the spacecraft for orientation and small translations in any direction. The baseline configuration is shown in Figure 4-1. A list of components is provided. The products of hypergolic combustion are directed opposite to required spacecraft motion and controlled by on-off valves. The thrust value of each rocket motor is small enough to allow fine attitude control by quick-acting valves. Timing for the precise attitude hold mode requires that a minimum impulse be applied at the required time to establish stabilization conditions for astronaut observation and full power communication.

The choice of an equipment configuration similar to Apollo will utilize already developed hardware in a suitably redundant condition to provide many alternate modes and gradual functional deterioration with increasing failures. With four equally spaced sets of four opposite pointing modes, all translations are possible — forward, backward, up, down, right, and left in addition to the turning moments of pitch-up, pitch-down, yaw-right, yaw-left, clockwise roll, and counter-clockwise roll.

DEFINITION OF DUTY CYCLES

Stabilization is required during zero-g periods of operation, which occur during earth orbit, during mid-course corrections en route to Mars and return, planetary encounter, and earth reentry orientation. Automatic, normal, and fine control, as well as astronaut capability for control in any direction will be possible. The detailed schedule for each mode has not been made because of the advisability of in-flight determination of specific times, but the reaction control subsystem is considered operating throughout the zero g times with capability for operation in any mode.

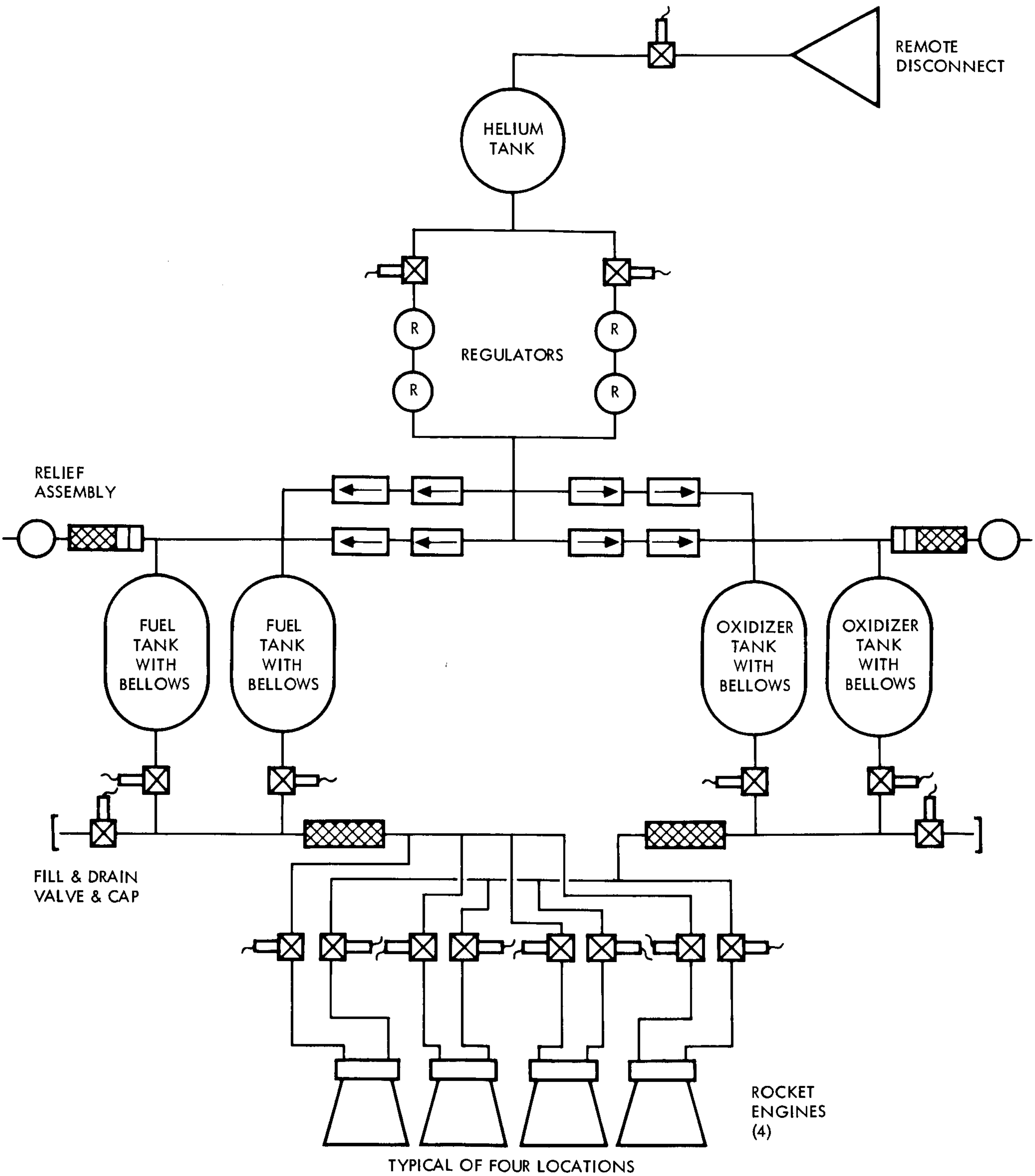

Figure 4-1. Reaction Control Subsystem Schematic

Every mid-course correction will require axial thrust for propellant acquisition and roll control during main engine firing. Although the main propellant tanks have passive propellant acquisition means (screens and baffles), the only position means is the aft-directed thrust of the reaction control subsystem. Roll control during thrusting is required for a single main engine, and may be relegated to a back-up capacity when multiple engines are used.

Docking maneuvers could occur at any time during zero g, but they are considered no more severe on the equipment than attitude hold. The capability of translation and rotation in any direction will be made available for immediately accessible astronaut implementation.

The normal method of reentry requires separation of the reentry vehicle from the rest of the spacecraft, and is accomplished by the operation of the forward thrusting engines. Once begun, separation should be as great as possible to make collision extremely improbable; i. e., retro thrust should be continued until propellant is exhausted. However, that requirement may be modified by specific vehicle design if the retro vehicle can be designed to burn and fall away safely or to utilize aerodynamics drag for positive separation. For this design, aerodynamic separation is considered as a back-up mode.

A preliminary duty cycle for the reaction control subsystem for attitude control, shown below, has been extracted from the overall mission timeline.

Phases 3.1 through 3.4	Propellant storage throughout 24 hours
	Engines tested for 100 cycles
Phase 3.5	Propellant storage throughout 48 hours
	Engines operated for docking stability for 150 cycles
	Engines operated for propellant acquisition and for roll control during thrusting, for 2 cycles
Phase 3.6	Propellant storage throughout 0.7 hours
Phase 3.7	Propellant storage throughout 3360 hours
	Engines operated to stabilize and to acquire propellants during zero g only and to provide roll control during thrusting for 602 cycles

Phase 3.8	Propellant storage throughout 20 hours
	Engines operated to stabilize and to acquire propellants during zero g only and to provide roll control during thrusting for 400 cycles
Phase 3.9	Propellant storage throughout 460 hours
	Engines operated to stabilize carefully during the planetary encounter for 8000 cycles
Phase 3.10	Propellant storage throughout 12,700 hours
	Engines operated to stabilize and to acquire propellant during zero g only and to provide roll control during thrusting for 602 cycles
Phase 3.11	Propellant storage throughout 20 hours
	Engines operated to stabilize during zero g only for accurate reentry corridor determination for 200 cycles
Phase 3.12	Propellant storage until separated from reentry capsule for 0.1 hours
	Engines operated to stabilize during zero g, to acquire propellants in the main subsystem, to provide roll control during retro, to stabilize after retro, and to propel spent vehicle away from reentry vehicle for 2 cycles.

All components will be subjected to vacuum environment throughout the flight, the engines external and exposed to meteoroid impingement and space radiation.

Reaction Control Subsystem Component List for Attitude Control

4 Apollo "Ouads"

Attitude Control Propulsion

4 Helium Tanks
4 Fill Valves and Disconnects
8 Helium Solenoid Valves
16 Regulators

8 Burst Discs and Filters
8 Relief Valves
8 Fuel Tanks
8 Oxidizer Tanks
16 Propellant Solenoid Valves
8 Main Filters
8 Fill Valves and Caps
16 Rocket Engines
32 Check Valves

IDENTIFICATION OF CRITICAL FUNCTIONS AND COMPONENTS

The reaction control subsystem satisfies critical functions with noncritical components. From the previous definitions of criticality (crew safety required, mission success required, and other), the following functions are crew safety and mission success critical:

1. Attitude control is required to provide orientation for midcourse correction impulses so that the earth reentry corridor will be achieved.

2. Roll control during impulse thrust is required to prevent incapacitation of astronauts and to utilize all main propellants. Uncontrolled rotation would prevent astronaut motion and move propellants away from the tank outlet.

The other three functions have alternate modes for their accomplishment:

1. Main propellant acquisition by internal screens and baffles.

2. Retro vehicle separation by aerodynamic force.

3. Docking maneuvers could be accomplished by the other vehicle.

All components are used in some redundant mode except during earth orbit. If unrepairable failure should occur in earth orbit, the mission would be aborted or another vehicle would be assembled. Once the craft is committed to the transplanetary mission, the redundancy is available to return the crew safely and to accomplish the mission.

Discussion of Redundancy Available

The attached reliability logic diagram (Figures 4-2 and 4-3) illustrates the multiple component redundancy available from the four reaction control modules. Each module is entirely separate from the

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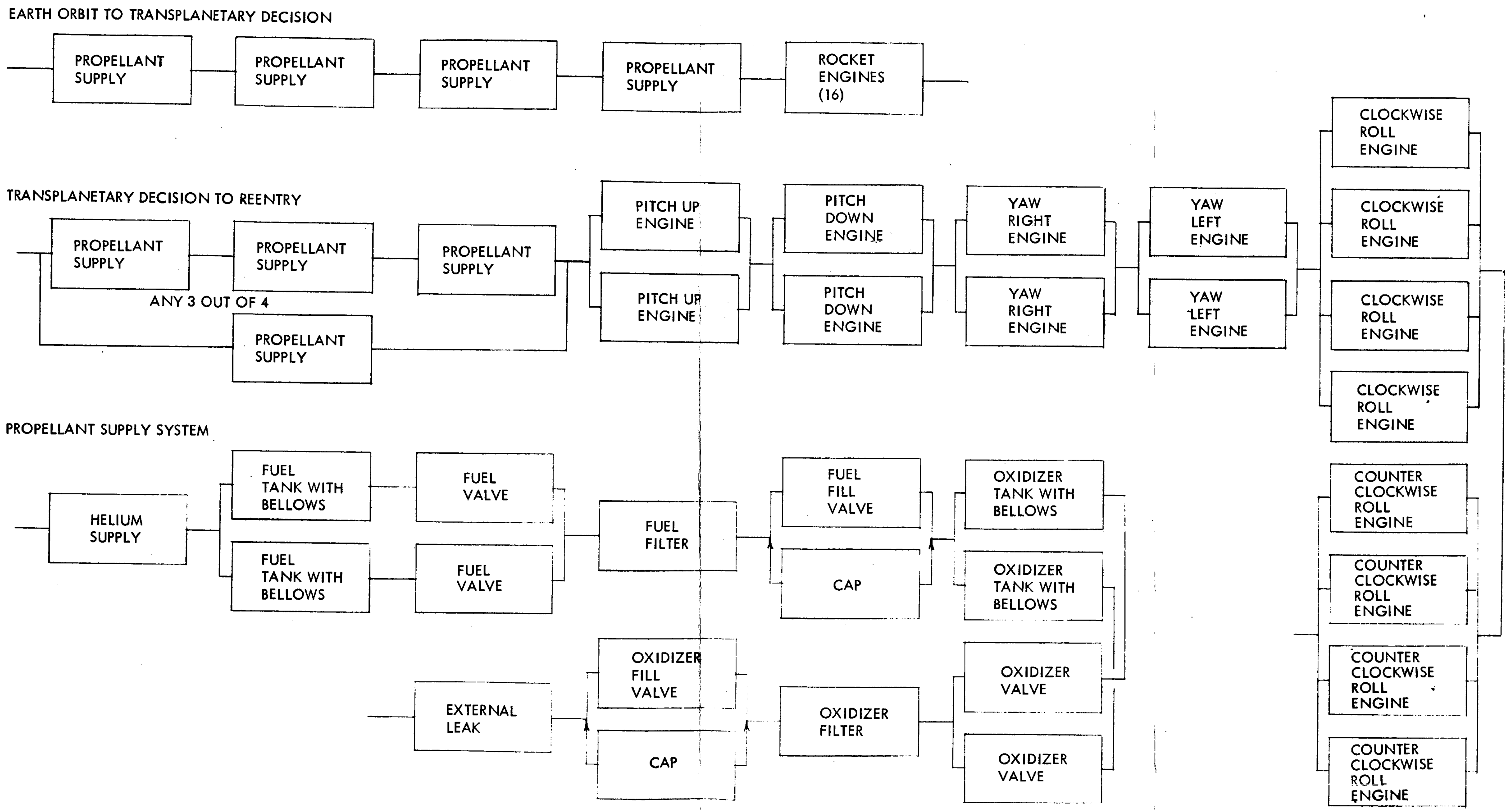
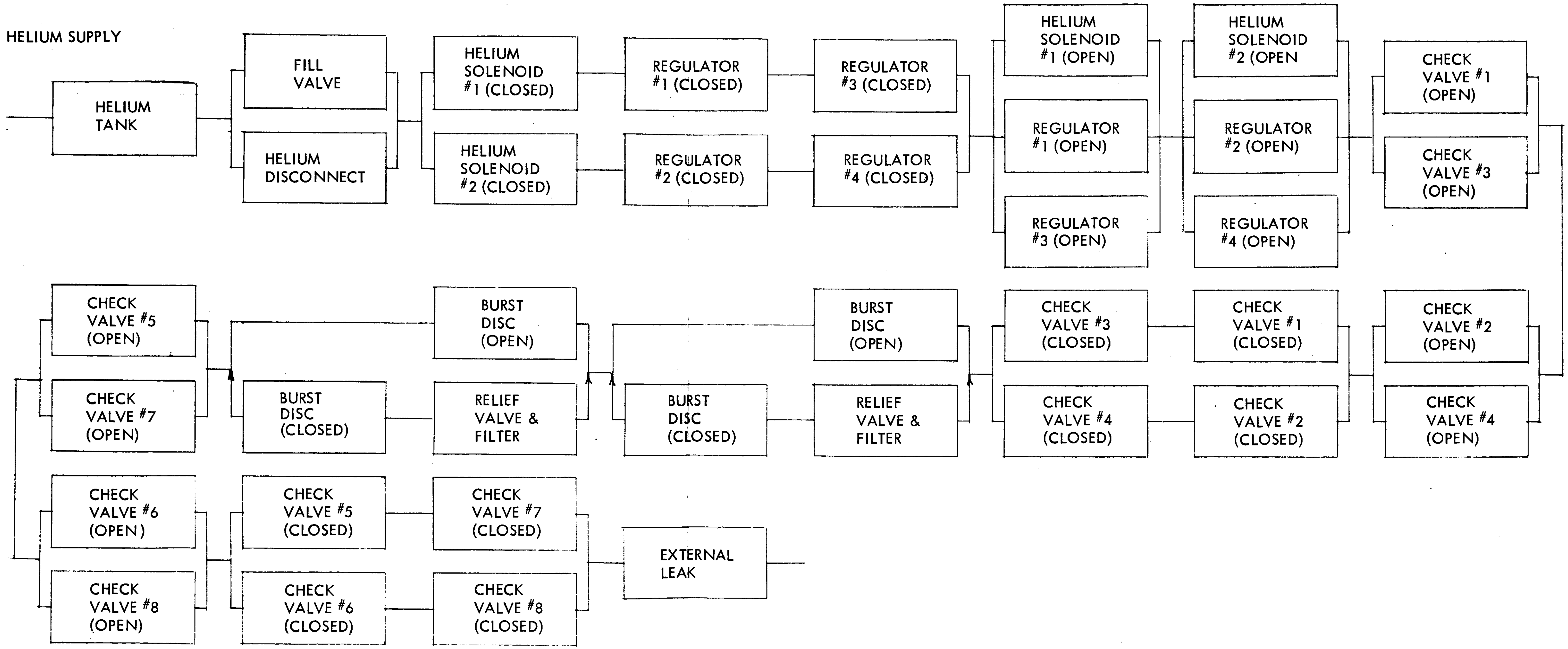


FOLDOUT FRAME 1

Figure 4-2. Reaction Control Subsystem - Reliability Logic Earth Orbit to Transplanetary Decision

FOLDOUT FRAME 2



FOLDOUT FRAME 1

Figure 4-3. Reaction Control Subsystem - Reliability Logic

FOLDOUT FRAME 2

others: no cross utilization of propellant is considered. However, not only are any three out of the four propellant supplies sufficient for crew safety but also each propellant tank within each module is considered redundant with the other tank within that module. The propellant fill and drain valves are not operated during flight. They serve only to retain the fluid and are backed-up by a cap, which is manually applied. The helium fill valve must be remotely operated because of launch area safety regulations when the tank is fully pressurized; therefore the quick disconnect must include a redundant seal mechanism. Helium control and regulation in each module are performed by a redundant array of two solenoid valves and four regulators. All three items in each leg (one solenoid and two regulators) have to be open if that leg is to operate successfully; but any one of the items can prevent over-pressurization from an open failure. Each set of four check valves is internally connected for redundancy in both forward and reverse flow. Preference has been given to prevention of reverse flow since uncontrolled passage of propellants would overpressurize the opposite tanks, even though slight reverse leakage may not be a severe problem; i. e., small leaks may increase tank pressure within acceptable limits. Since the relief function should never be used in flight, a burst disc will maintain a positive seal. This will be backed up by a re-setting relief valve.

Rocket engines are required to provide the attitude control thrust necessary for spacecraft return. As shown in the logic diagram, all attitude functions are accomplished redundantly with roll in either direction by any one of four engines. The representation of engine redundancy is difficult on a logic diagram, since the indicated logic only occurs when all propellant supplies are operative. When one propellant supply fails, four engines are inoperative, and redundancy is lost in either pitch-up and pitch-down or yaw-right and yaw-left. This slight decrease in reliability is considered compensated for by the possibility of continued, though difficult, attitude control with two modules.

Discussion of Critical, Low Reliability Assemblies

The limited use during zero g and translation allows reasonable, high reliability results from components not much better than exist today. However, difficulties exist in the design of this subsystem, even though there are neither critical components nor low reliability subassemblies. As shown in the spin engine section, helium is difficult to retain during long flights, the bellows method of positive propellant expulsion weighs much more than a comparable bladder, and the extent of meteroroid flux on the exposed engines is relatively unknown. Therfore, recommendations are again made to develop auxiliary means for propellant pressurization, to continue development of an impermeable bladder, and to enclose the engines in a meteoroid bumper during the spin mode.

Redundancy Required

Existing redundancy is necessary for the functions required; however, if the meteoroid bumpers should be impractical, spare rocket engines would provide extra assurance. A redesign of the engine connections could allow quick disconnect and replacement by an astronaut in extra-vehicular maintenance.

SPECIAL CONSIDERATIONS

The only interfaces between the attitude control subsystem and the others (except for structures) is in the electrical control. With an adequate electrical power supply, no serious problems are anticipated from plume impingement or simultaneous separation of guidance, communication, and main propulsion subsystems. Batteries may be used to provide peak power for the gimbal actuators during mid-course correction. Optical and thermal transmission and reflection may be affected by the products of combustion, but this remains to be proven.

Development of reliable equipment for the previous recommendations should begin at once; so that the best design can be made at the time of design freeze. Even though the bellows expulsion is available, the weight expense should be minimized. More accurate determination of meteoroid flux must be accomplished for minimum weight bumper design. The effects of engine exhaust plume will be adequately documented on the many Apollo flights already planned. Helium storage provisions for long duration should be tested and developed to insure minimum leakage. Investigation of alternate pressure sources should continue as a parallel effect. Finally, seal materials should be tested under long exposure to propellants and with multiple exposures to space environment. These developments are considered the most fruitful for increasing performance and reliability.

Space probes, earth orbit vehicles, and earth-based multiple environment chambers are the required additions to the present facilities. The few probes will cover expected manned vehicle paths, and a single, earth-orbiting laboratory will satisfy space duration requirements. Earth-based chambers should be designed to provide simulated radiation, vibration, vacuum, and stress, so that the effects can be observed before actual space operation. The earth-orbit program could replace the earth-based laboratory if a timely development program is scheduled.

RESULTS OF SUBSYSTEM ANALYSIS

Success of the entire mission depends on the operation of the reaction control subsystem. It is imperative that the attitude control function be accomplished during every zero-g operational phase. Utilization of Apollo equipment (with some additional development) will build on current technology toward the new interplanetary environmental and performance requirements.

Recomp computer limitations prevent exact computation of the probability of subsystem success from the detailed reliability logic and estimated failure rates. Since, a quick assessment of the reliability indicates a value greater than 99 percent, the problems are considered to be at the component level rather than in the configuration. If an increase in the reliability goal is required, another review of the entire subsystem will be necessary.

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5. MAIN ENGINE SUBSYSTEM

FUNCTION

Major translation impulses will be required for transplanet and transearth midcourse corrections, planet encounter trajectory adjustment, and earth reentry corridor acquisition. To accomplish this with the transplanet injection booster, exceptional accuracy beyond the present and near future state-of-the-art would be required. The many extraneous forces integrated over a mission of long duration cause sizable errors in spacecraft location direction, and velocity; e.g., multiple-body gravitational effects, solar wind pressures, accuracy of navigation equipment, and injection conditions. Early correction will remove most of the injection errors, and subsequent small impulses will improve the accuracy of planetary encounter and reentry corridor acquisition.

Dependence on a single engine, (as shown in Figures 5-1 and 2), to make these critical corrections entails a severe risk, even though redundant equipment items are placed throughout the subsystem. It is difficult to make the several critical items as reliable as desired, and the risk of the effects of single failure is unnecessary.

With one large single engine, timing of the impulses is very important so that the corrections are made as accurate as dictated by the guidance requirements, especially for planet encounter and reentry trajectories.

DEFINITION OF DUTY CYCLES

After the initial test firing in earth orbit, the present plan is to make three midcourse corrections during transplanet trajectory and three during transearth trajectory, with the last firing utilizing all remaining propellants to decrease reentry velocity. The firing times and numbers of operations are appoximate, since each trajectory will be affected by different forces and will require different corrections. The example duty cycle for the main propulsion subsystem is as follows.

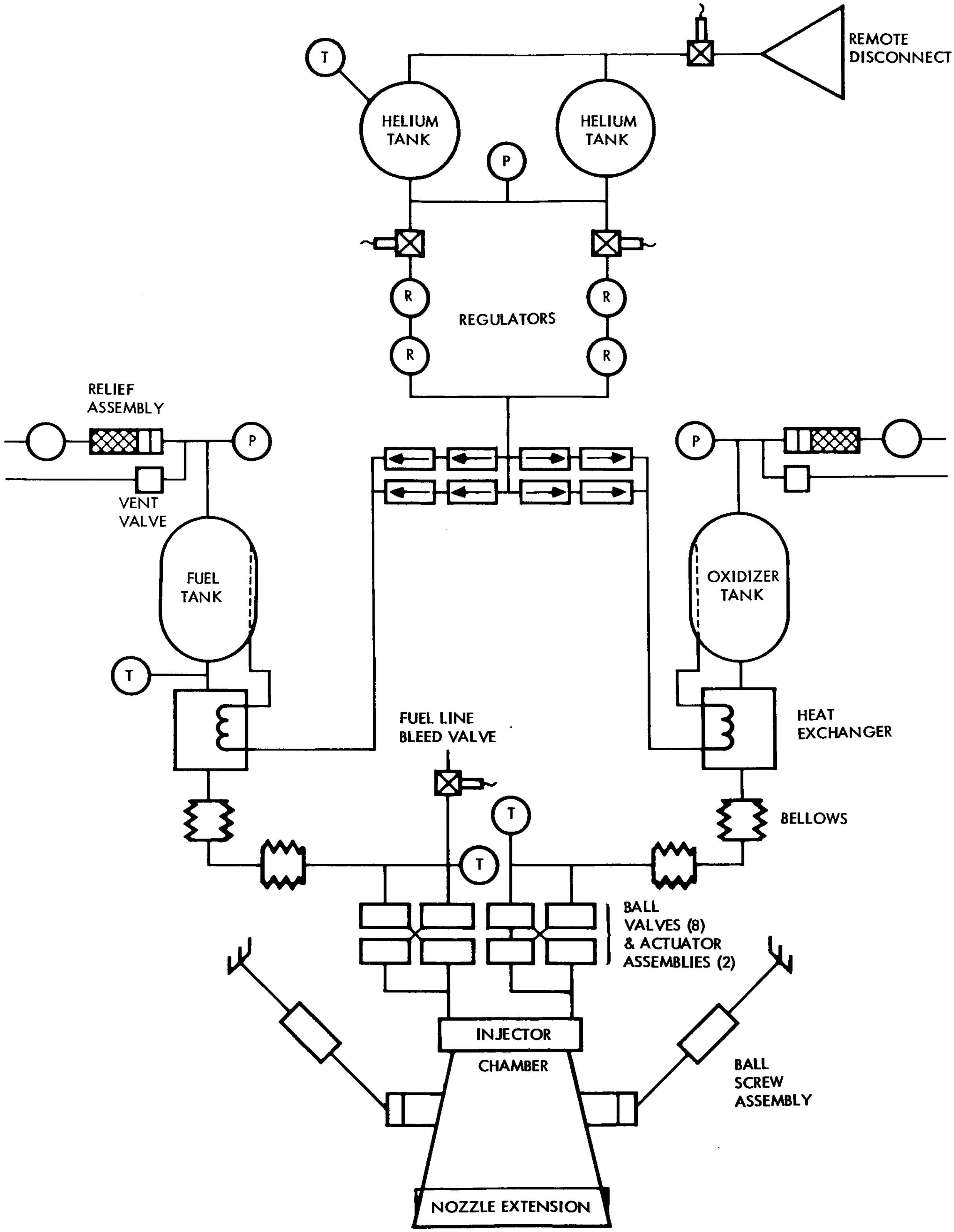

Figure 5-1. Main Propulsion Subsystem Schematic

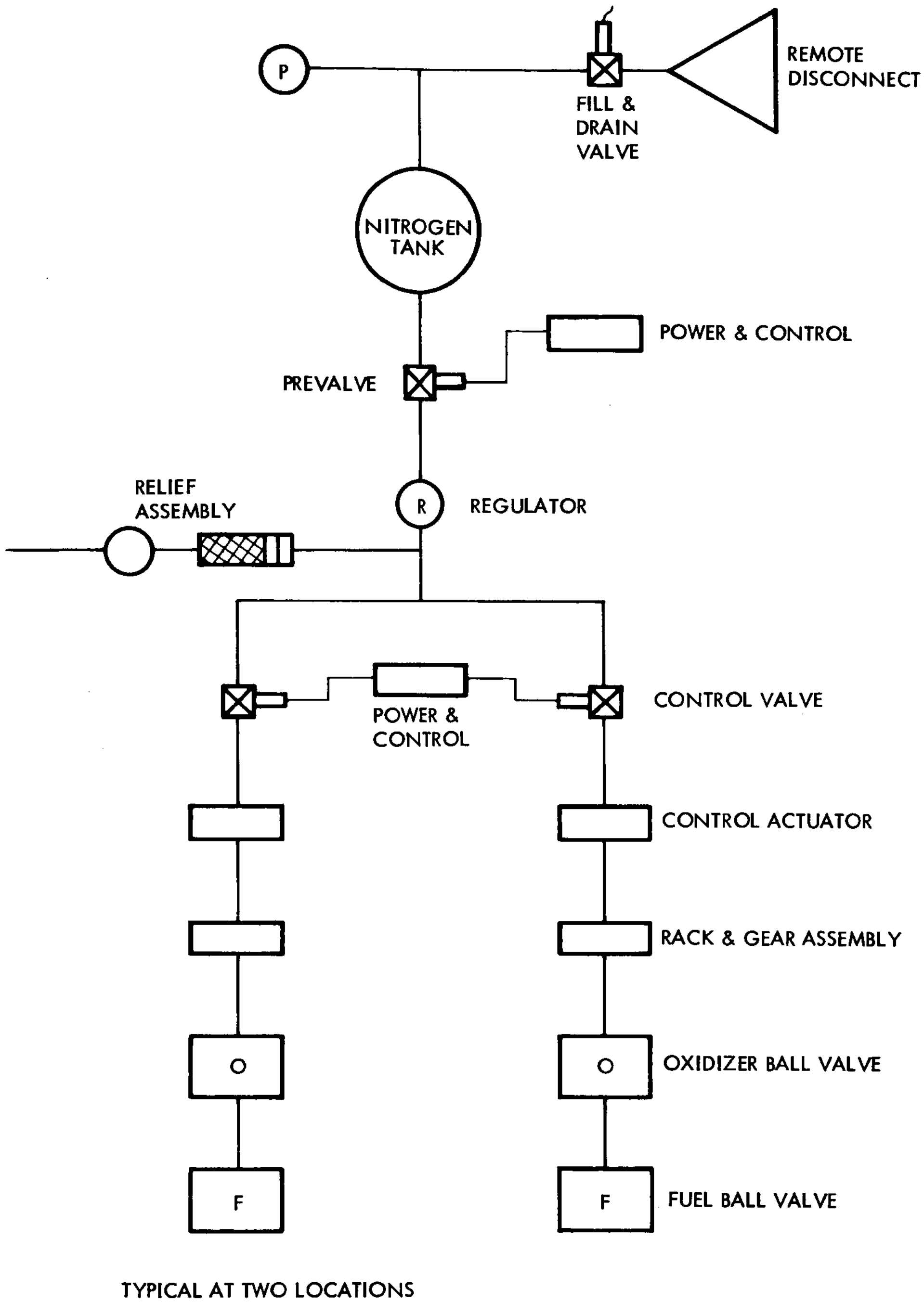

Figure 5-2. Ball Valve and Actuator Assembly Schematic

Main Propulsion Subsystem for Course Correction

Phases 3.1 through 3.4	Propellant storage throughout 24 hours
Phase 3.5	Propellant storage throughout 48 hours
	Engine operated for one test firing of 0.5 sec. duration
Phase 3.6	Propellant storage throughout 0.7 sec.
Phase 3.7	Propellant storage throughout 3360 hours
	Engine operated for one course correction for 25 sec.
Phase 3.8	Propellant storage throughout 20 hours
	Engine operated for two course corrections to obtain correct planetary encounter trajectory for 50 seconds
Phase 3.9	Propellant storage throughout 460 hours
Phase 3.10	Propellant storage throughout 12,700 hours
	Engine operated for two midcourse corrections for 50 sec.
Phase 3.11	Propellant storage throughout 20 hours
Phase 3.12	Propellant storage throughout 0.1 hours
	Engine operated for one retro firing for 25 sec.

All parts of the engine will be exposed to space environments throughout the mission. A meteoroid bumper is anticipated around the propellants but not around the thrust chamber, gimbal equipment, and nozzle.

Main Propulsion Subsystem for Course Correction (Subsystem Component List)

Propellant Supply

2 Helium Tanks

1 Helium Fill Valve and Cap

1 Helium Fill Coupling

1 Helium Temperature Transducer

3 Helium Pressure Transducers

2 Helium Solenoid Valve

4 Helium Regulators

2 Circuit Breakers, Switches, and Relays

9 Test Points

8 Check Valves

2 Heat Exchangers

2 Propellant Vent Quick-Disconnect Assemblies

2 Relief Valves and Burst Discs

1 Bleed Valve and Test Point

2 Propellant Fill and Drain Assemblies

4 Propellant Tanks

3 Propellant Temperature Transducers

2 Flexible Bellows

Engine Assembly and Gimbal Actuators

1 Engine Assembly

2 Ball Screw Assemblies

2 Bellows and Cases

4 Gimbal and Bearing Assemblies

1 Nozzle Extension

1 Chamber Pressure Transducer

2 Ball Valves and Actuation Assemblies

IDENTIFICATION OF CRITICAL FUNCTIONS AND COMPONENTS

Provision of impulse for course correction is required for mission success and crew safety. Acquisition of planet encounter and reentry trajectories is dependent on this subsystem to progressively correct the error caused by the multiple environments and inaccuracies. The function of retro thrust at reentry is considered to be redundant with the reentry heat shield, since the effect of retro thrust will be to increase the safety margin of the shield but not to bring the velocity within the shield capability.

Components are also critical to the mission and to the crew. Even though a valiant attempt has been made to provide internal redundancy, success of the current subsystem is unnecessarily dependent on single failure modes, especially for long duration missions; e. g. , helium supply tank, heat exchangers, propellant tanks, bellows, fuel line bleed valve, portions of the gimbal actuator, and the injector, thrust chamber, and nozzle assembly.

Redundancy Available

Much internal redundancy is available, as shown by the following reliability logic diagrams, Figures 5-3 and 5-4.

1. Propellant fill valve seals are redundant with the manually placed cap.

2. Helium fill valve requires a quick-disconnect back-up because of pad safety precautions at full tank pressure.

3. The control solenoid and regulator array is similar to the attitude control configuration and provides redundancy in either fail-open or closed.

4. Check valve arrays provide redundancy in the forward and reverse flow control with emphasis on prevention of reverse flow. Uncontrolled interchange of propellants would over-pressurize the opposite tanks.

5. Vent valve seals are redundant with quick disconnects.

6. The relief function is not expected to occur; therefore, the relief valve is in standby redundancy with the positive seal burst disc.

7. Nitrogen gas actuation assemblies for the main propellant valves are redundant; i. e. , one assembly is in stand-by, while the other controls the operating ball valves.

8. By ingenious arrangement, the main propellant ball valves are redundant for failure open, closed, and internal leakage.

9. Propellant line bellows have several plies of material; so several layers of internal redundancy.

10. Selective internal portions of the gimbal actuator assembly; e. g. , motor, electric clutch, travel indicators, and rate sensors.

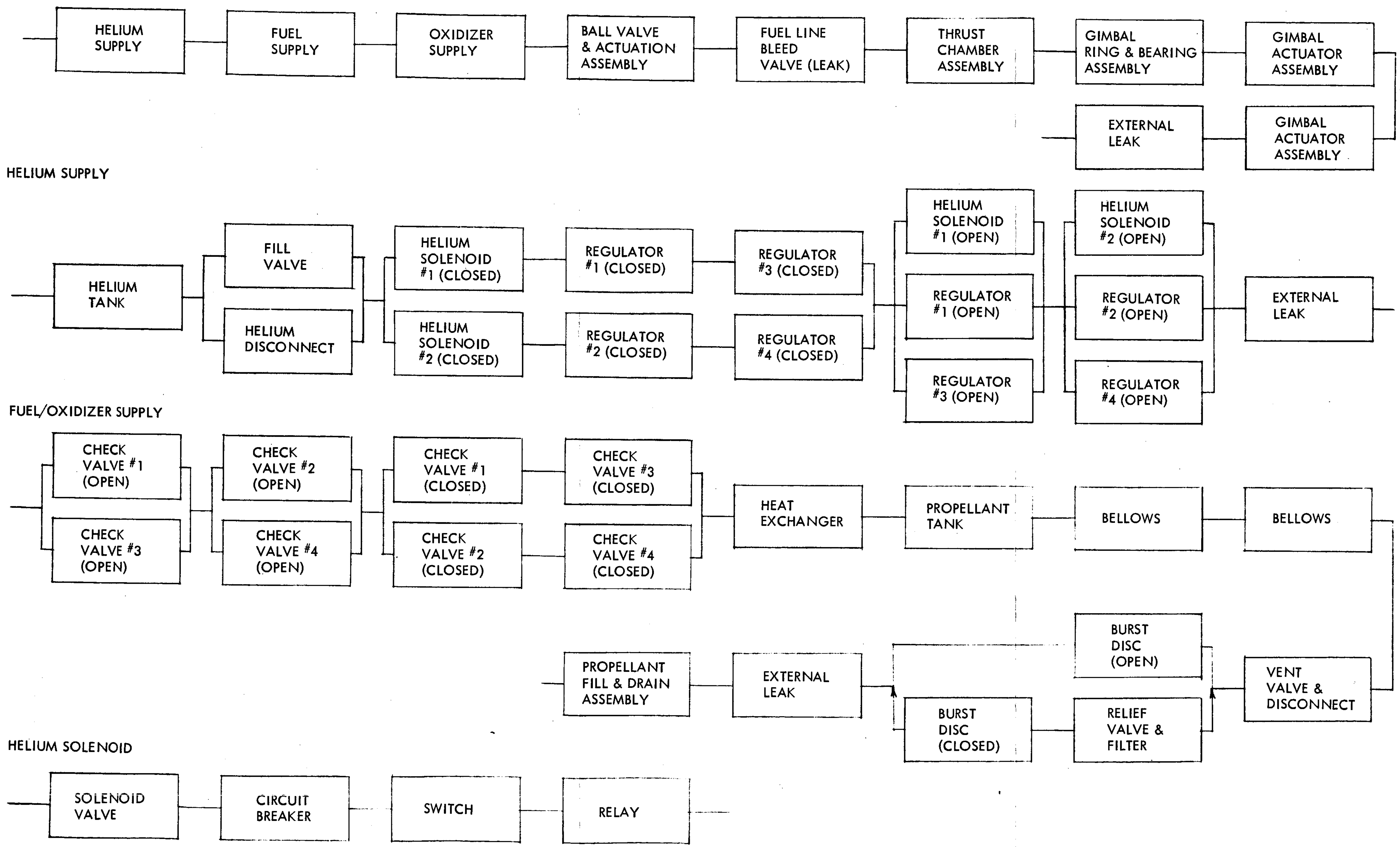

Figure 5-3. Main Propulsion Subsystem - Reliability Logic

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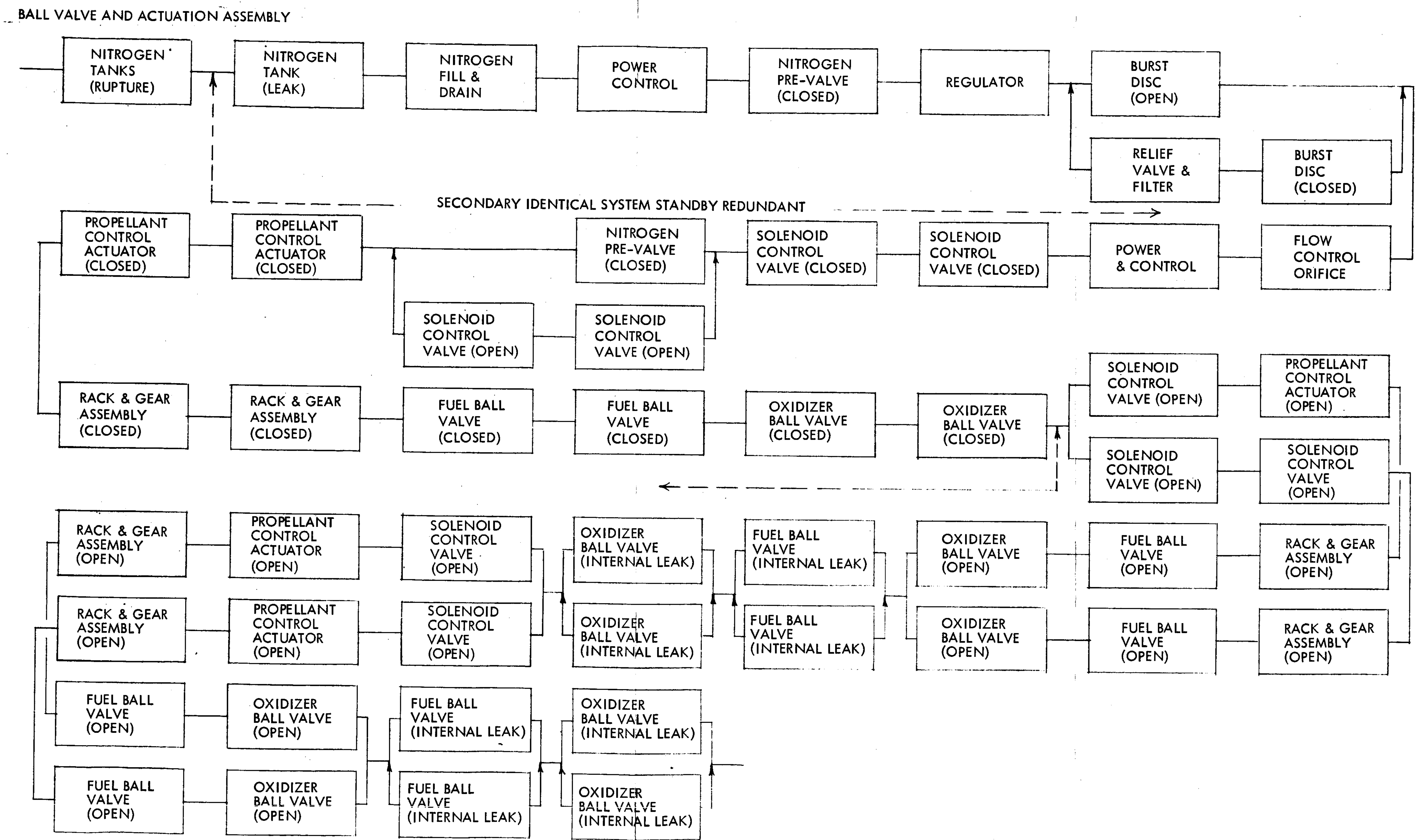

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Figure 5-4. Main Propulsion Subsystem - Reliability Logic

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Discussion of Critical, Low Reliability Assemblies

Many critical components required for mission success and crew safety have been designed with a large safety margin to lower the risk as far as practicable.

Helium Tank

Helium is very difficult to retain within tank materials of current types. The large margin required to make the component sufficiently reliable also increases the weight. Continued development of helium containment materials is required to minimize penalties of long-term storage. Continued development of an alternate pressurization means would provide needed redundancy, as illustrated in the Spin Engine Section.

Heat Exchangers

Although the heat exchanger size is large enough to provide the pressurizing gas heating function, any internal leakage reduces that margin, and external leakage is critical. Reliability is aided when there are no moving parts, but careful design is required to account for all the internal stresses from differential temperatures.

Propellant Tanks

The safety margin seems adequate at the present time, but will require review as operating experience is gained.

Fuel Line Bleed Valve

No redundancy now exists for this valve, whose function is to insure propellant at the ball valve inlet by bleeding gas from the line. If no in-flight actuation is required, a manual cap could be placed over the end of the line.

Gimbal Actuators

Portions of the actuator are not redundant; e. g., structural rods, ball screw assembly and bearings, and internal gearing. Another critical review of this component is required, since the current reliability goal has not been demonstrated.

Injector and Chamber

That these are taken together signifies their interdependence for combustion stability and low heat transfer rate. Traditional development has been extensive to achieve desired reliability. Current demonstrated reliability is insufficient to accomplish the multiple starts and short runs required. Even though the duration margin is very large, the frequency of incidence of combustion instability is still too high. A possible solution is replacement by three assemblies of lower thrust, to provide redundancy, to allow greater accuracy in impulse application, and to take advantage of a shorter profile for low weight meteoroid bumpers. Exposure of the engine to long coast periods of meteoroid bombardment is a large risk at the present maximum flux.

Ball Valve Seals

Even though redundant, the sealing capabilities of the ball valves are questionable over long periods of times in contact with propellant. Not only is compatibility unproven, but also seal deformation with time reduces its effectiveness. Careful monitoring of current design changes to a poppet type of seating arrangement and inclusion of additional valving to increase seal redundancy must be made before the function can be considered capable of high reliability accomplishment.

SPECIAL CONSIDERATIONS

Interfaces with other subsystems are important considerations; e.g., guidance control of the gimbal actuators, heating of the rear of the spacecraft from the jet expansion around the nozzle, cut-off control for accurate impulse achievement, and environmental control to prevent propellant freezing.

If begun now, development of all the recommended assembly changes is possible within the expected time period before an interplanetary mission. In particular, the integration of multiple rocket engines with the propellant tanks and structure should be commenced as soon as hardware is available. This would be beneficial not only to the interplanet vehicles, but also to the current Apollo.

The only special facilities required would be the results of probes for meteoroid data and an earth orbit, long duration, seal laboratory.

RESULTS OF SUBSYSTEM ANALYSIS

Even though much effort has been placed on internal redundancy, the remaining critical items are numerous and full of risk. A detailed review should be undertaken, and a long-term development test plan prepared. Many changes may be required before the subsystem is considered adequate for the increased severity of the interplanetary mission.

The large number of success modes considered in the reliability logic diagram exceeds the capability of the Recomp computer. An approximate value of reliability was determined to be 90 percent based on expected component increases. Both the configuration and the individual component contribute to this reliability low value; therefore, both could be improved for the next design.

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6. ELECTRICAL POWER SYSTEM

FUNCTION

The selected electric power system consisting of a radioisotopic heat source for a primary dynamic Rankine cycle with backup thermoelectric conversion provides flexibility to meet mission requirements and contingencies. The manned mission has as a minimum requirement the need for sufficient electrical power to support life while permitting spacecraft operation during return under emergency conditions. If an electrical power contingency develops shortly after mission commitment, minimum backup electrical power of approximately 3.6 KWe must be available for well over one year for a four-man environmental system and other essential spacecraft operations.

OPERATING REQUIREMENTS

The electric power system must provide power for the entire mission duration, as noted on the timeline of Section II.

IDENTIFICATION OF CRITICAL FUNCTIONS AND COMPONENTS

A continuous source of electrical power is essential to both mission success and crew safety; however, with the exception of the isotope source and boiler, redundancy is provided throughout the system. The estimated reliability of the isotope and boiler is extremely high, and the characteristics of this component are such that redundancy is not feasible in any event.

Available Redundancy

The primary dynamic Rankine mercury system provides for the operation of any two of a total of seven available 3 KWe units. A cascade thermoelectric system will permit a 3.6 KWe output for backup power or emergency conditions in the event of complete primary system failure. Figure 6-1 represents a reliability block diagram of the system.

Design considerations affecting reliability of the isotope electric power system include high integrity of the fluid circulating loops combined with the necessity of continuous heat removal from the radioisotope boiler from inception through final disposition. The single radioisotope boiler system is to be designed for intact return to earth. Heat removal is effected by CRU, thermoelectric and/or excess isotope power rejection systems.

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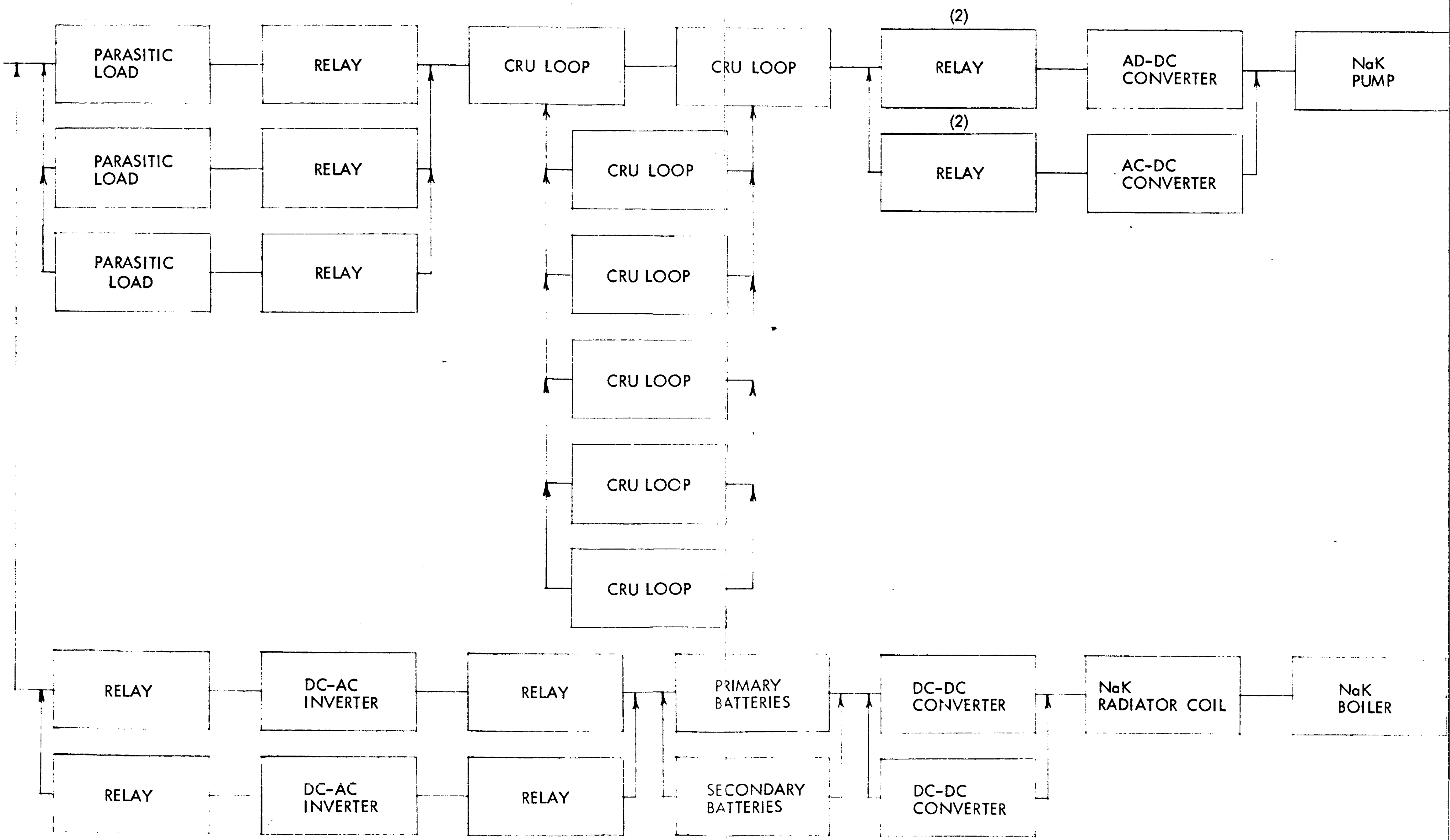

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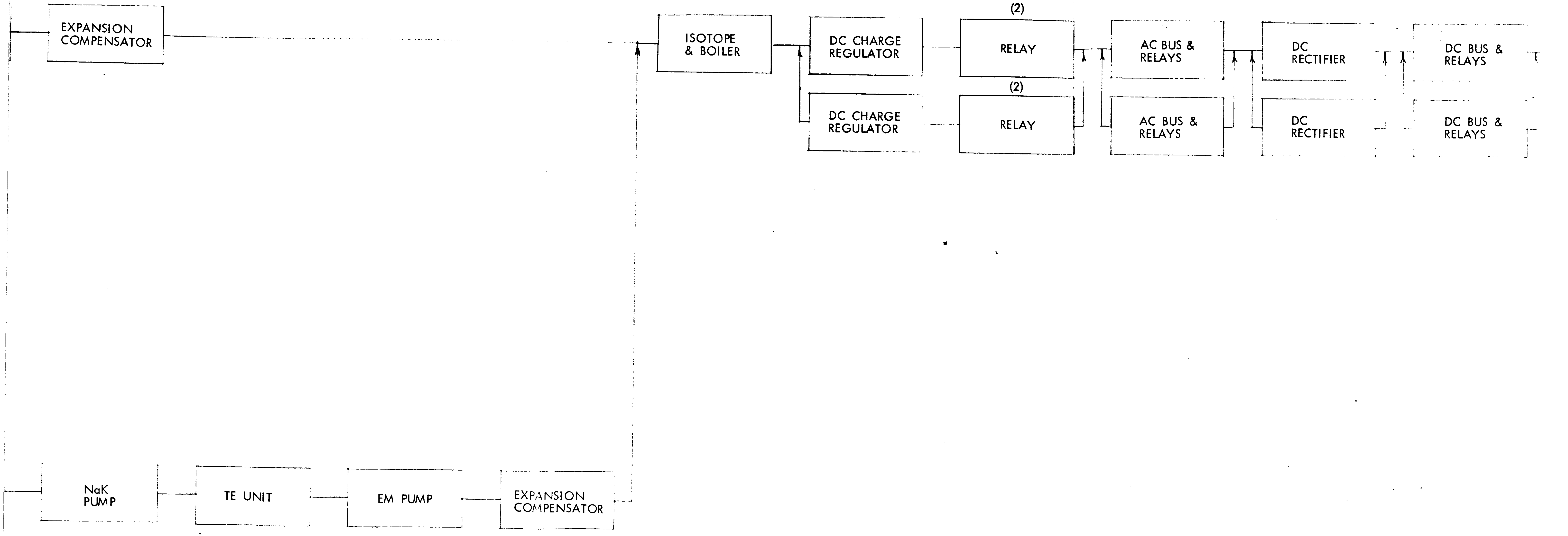

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Figure 6-1. Electrical Power System

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Since leakage in any loop is expected to inactivate the associated loop equipment, two separate loops in parallel will provide added assurance against leakage failure of the thermoelectric and the excess power rejection systems. Each of the seven CRU systems is to be provided with a separate single circulating system to minimize loss of the primary power system capability by fluid leakage. The above combination of pumped fluid systems provides a choice of continuous heat removal. Excess heat removal by CRU system operation will be made at the expense of added CRU operating time and/or capacity to reject the electrical energy as heat from parasitic load resistors.

The nature of eutectic NaK in the excess power and thermoelectric loops and in the primary CRU loops requires special consideration of their presence outside the system. Long-term human exposure to mercury vapor at normal temperatures is undesirable, and its effect on sensitive equipment can be harmful to its operation. NaK is readily oxidized in the presence of O_2 to the extent of being considered incendiary when hot or exposed to water, or its vapor, with the attendant production of H_2. The oxidized products of NaK are caustic and irritating to human tissue and mucous membranes. For these reasons, fluid lines containing these materials will be isolated to avoid their introduction into sensitive system compartments, including the life support system. Remote control and monitoring features will permit suitable isolation on the spacecraft.

The dynamic isotopic Rankine mercury system utilizes a two phase liquid vapor operation in both the spin-up and zero-g conditions. While ground tests and studies indicate that the system is suitable for zero-g conditions, confirmation of two-phase static condenser designs is recommended during AAP flights.

A battery assembly of primary and secondary batteries is provided to supplement the primary power systems during peak loads for limited duration, if required. The batteries are rated at 150 AH/4200 WH. Reentry batteries of 12 AH/3360 WH are provided to furnish power required for reentry and recovery operations.

Additional backup consisting of a solar voltaic system was evaluated.

The static solar voltaic system has the longest history of successful power generation for space systems. Its use as a supplementary and/or emergency system in flyby missions is justified by demonstrated reliability in space. The present uncertainties of meteoroid effects for proposed mission exposure times and trajectories related to sizing and design will be reduced by future probes. In addition, continued advancements in the development of solar cell capabilities are expected to extend effective outputs and further reduce need for reserve exposure areas to compensate for space environment damage.

However, because of complications introduced by panel orientation requirements (particularly during the spacecraft spin mode) and dependability of the basic system, this system was not considered for purposes of the interim report.

Critical, Low Reliability Assemblies

Utilizing failure rates provided by Atomics International, a reliability estimate of 0.983 for the entire mission was calculated. This compares with the value of 0.969 obtained by computer simulation. The elements possessing the highest failure rates (λ) are; (1) ac-dc converter with a failure rate (AI) of 25×10^{-6}; (2) dc-dc converter, $\lambda = 25 \times 10^{-6}$ (estimated); (3) expansion compensator, λ 3.6×10^{-6}; and (4) CRU's, $\lambda = 6.686 \times 10^{-6}$. The CRU's contribution to the system unreliability is negligible because of the number of stand-by units. The converters not only have stand-by units, but are in separate loops; isotope dynamic and emergency dc. The expansion compensator has no stand-by, but one unit is in each of the two systems. The contribution of these elements to the total subsystem unreliability of 0.017 is as follows:

Expansion Compensator	9.4 percent
ac-dc Converter dc-dc Converter	14.2 percent
Isotope and Boiler	5.0 percent

The remainder of the unreliability is a function of a number of items posessing fairly low failure rates and the extremely long operating time. No other single unit or assembly contributes as much unreliability as the items listed. However, it is probable that design improvements, additional test data, and reliability improvements which can be expected to occur prior to vehicle fabrication will result in overall reliability improvement.

Redundancy Required

No additional redundancy is considered feasible or necessary.

SPECIAL CONSIDERATIONS

Considerable development work is required for the isotope dynamic systems in general. No space missions utilizing these systems have been accomplished. However, ground test results conducted to date indicate high inherent reliability, which will be validated by flight experience prior to the Manned Mars mission.

7. ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

FUNCTION

The function of the Environmental Control/Life Support System (EC/LSS) is to provide a controlled and safe environment relevant to temperature, humidity, and atmospheric pressure and composition, remove all wastes, including CO_2 and liquid, and reclaim oxygen and water. The subsystem as currently defined includes a cryogenic storage for oxygen and nitrogen, a water reclamation system, and a CO_2 removal and O_2 recovery system. The system is largely redundant, which will permit shutdown for repairs or preventive maintenance.

OPERATING REQUIREMENTS

The EC/LSS must operate continuously throughout the mission, either totally or through one of the several redundant loops, as indicated in Figure 7-1. Total operating hours are equivalent to mission length, as defined in Section 2.

IDENTIFICATION OF CRITICAL FUNCTIONS AND COMPONENTS

All functions performed by the EC/LSS system are essential to crew safety, although the crew could function in an environment which drifts beyond the upper and lower limit of atmospheric composition and temperature by a small amount. While no suit circuit is provided as an alternate operating mode, the suit circuit itself is dependent on the basic system, and it would not contribute to crew safety for a long mission except in the event of a decompression caused by a repairable penetration of the pressure structure.

Available Redundancy

A logic diagram representing the EC/LSS system is represented by Figures 7-1 and 7-2. The thermal control, humidity, and air temperature control and CO_2 reduction/electrolysis oxygen recovery units are redundant and **are** shown on the diagram as redundant loops. Not shown in the diagram, are multiple tanks for N_2 and O_2; however, these tanks cannot be considered truly redundant, because the mission probably could not be completed if appreciable amounts of N_2 and O_2 were made unavailable.

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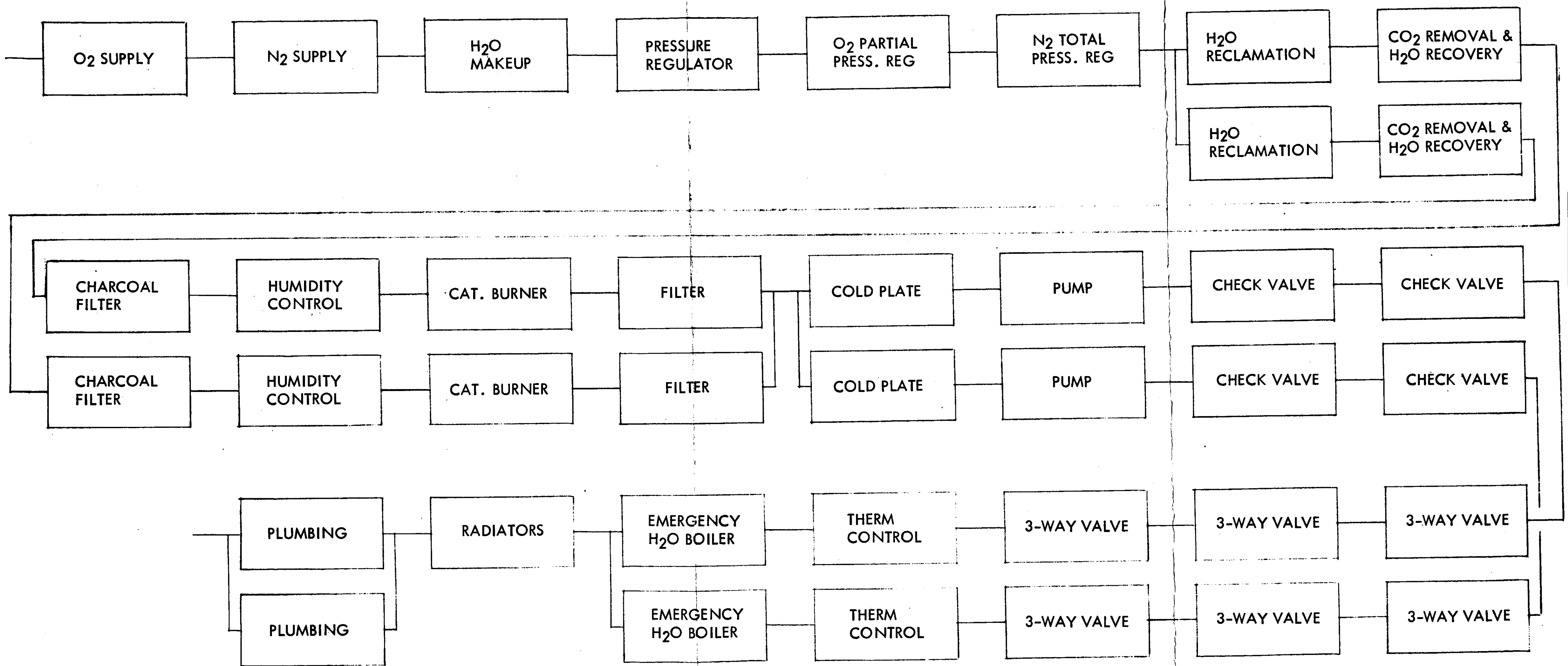

Figure 7-1. Environmental Control System

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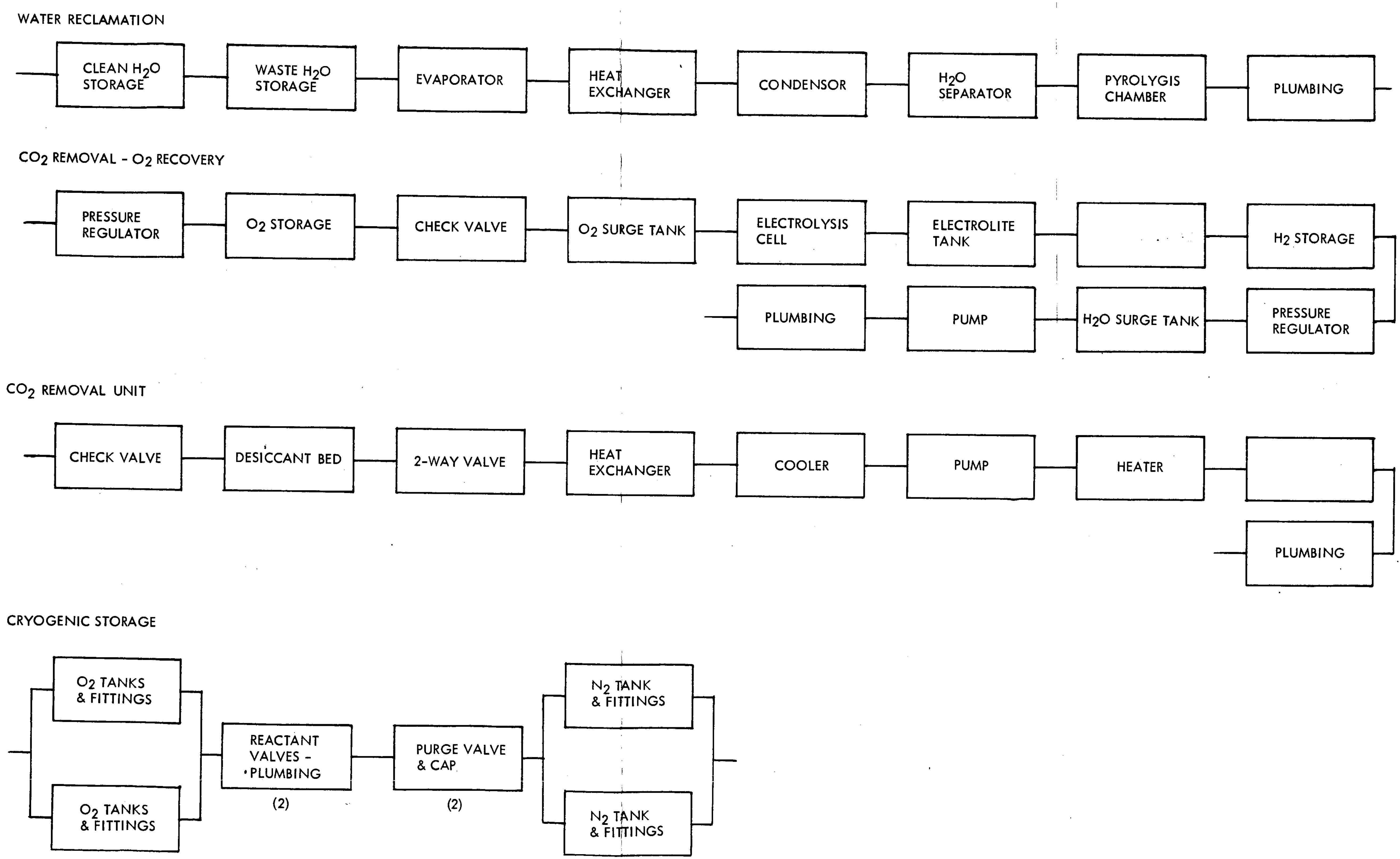

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Figure 7-2. Environmental Control System

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Critical, Low-Reliability Assemblies

Failure rates derived from SID 65-761, Manned Mars and/or Venus Flyby Vehicle Systems Study and from the Apollo Applications Program were applied to the logic diagram blocks to obtain an indication of system reliability. The estimates so obtained provide a gross indication of reliability performance and serve to pinpoint those areas contributing most to unreliability.

The preliminary estimate obtained by correlation of operating requirements and failure rates mentioned above indicates a reliability of 0.855, which is inadequate for a manned mission. The cryogenic storage contributes about 45 percent to the unreliability of the system. However, the design of the N_2/O_2 source has not been defined. Therefore, for purposes of preliminary analysis the cryogenic storage of Block II Apollo was used, with the H_2 tanks being replaced by N_2 tanks. This system was not designed for an extremely long mission and is perhaps more complex than required because of its added function of supplying the Apollo fuel cells. Design simplification will be the subject of additional study.

For the remainder of the system, no one assembly stands out as a major source of unreliability, with the exception of the redundant pumps which contribute about 16 percent to system unreliability.

Design refinements during the next study phase will improve the reliability; however, it seems certain that inherent reliability must be augmented by maintenance, as described in SID 65-761, to provide an acceptable degree of crew safety. Evaluation of specific items for which maintenance action or spares are recommended will be included in future study activity.

SPECIAL CONSIDERATIONS

While it is recognized that the description and schematic layout are preliminary, several aspects of the system should be clarified:

1. The method for returning the air from the conditioner compartments to the environmental system is not indicated.

2. The method by which the humidity control removes condensate is not clearly defined. If it is done by absorbtion, the reactivation process is not shown. If the humidity control is in the cold glycol loop having an apparatus dew point temperature corresponding to the glycol temperature, provisions are required for transporting the condensate from the humidity control to the H_2O recovery system.

3. Provision must be made for glycol circulation during the heating cycle when the radiators are not in use.

4. Crew safety aspects during depressurization of both compartments should be evaluated.

5. In event of emergency, when radiators are inactive or damaged and the water boiler is in use, provisions are necessary for glycol circulation.

RESULTS OF SUBSYSTEM ANALYSIS

Analysis results indicate that crew safety considerations require improvements which probably cannot be achieved in the near future by increases in equipment reliability alone. Provisions for maintenance must be incorporated to provide adequate assurance of crew survival.

This subsystem will require considerable development prior to mission use. This country has not as yet developed a closed loop system having actual mission experience. This subsystem must be categorized as one of the most critical equipment items, considering the current development status and the essential nature of the function.

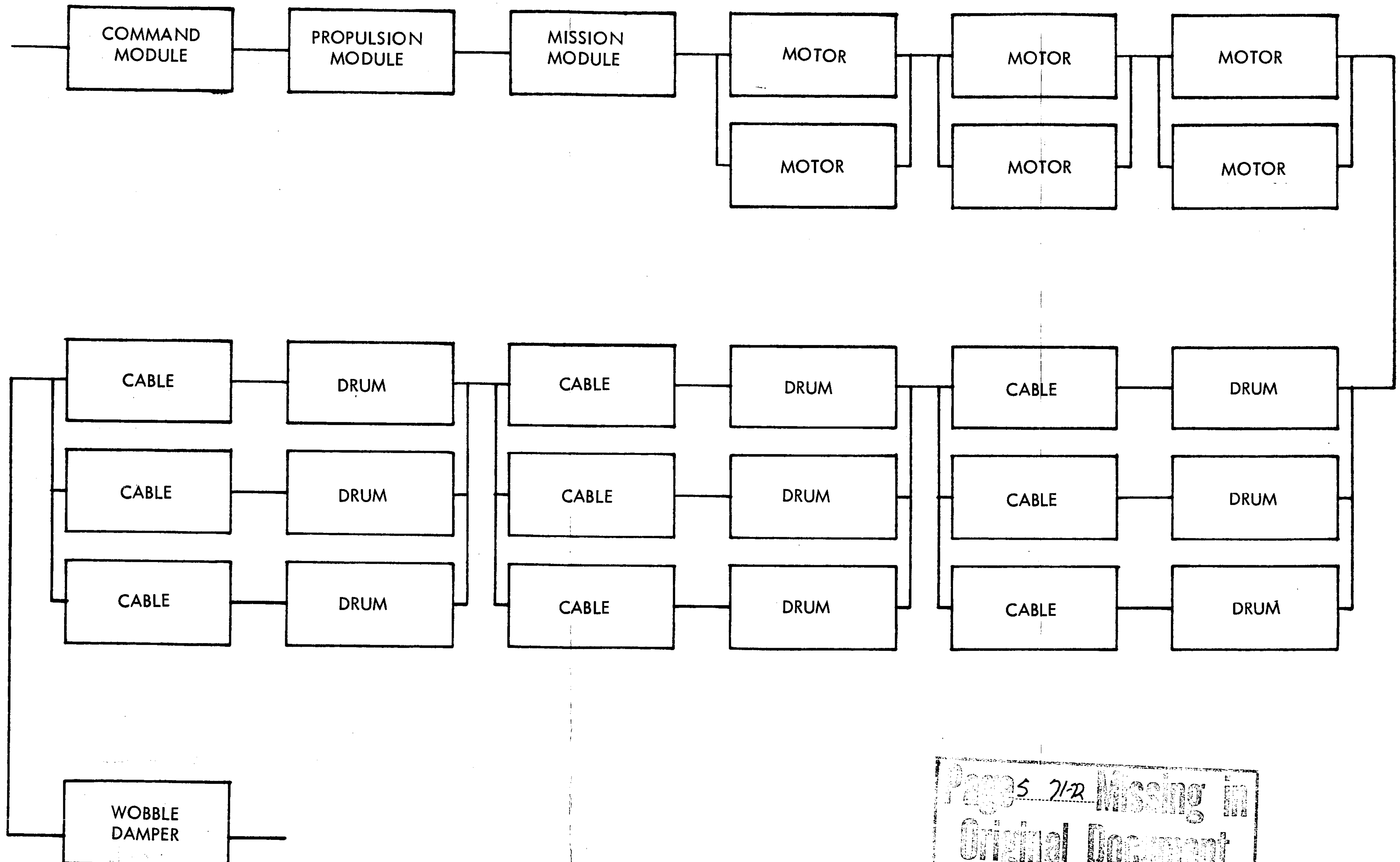

Figure 8-1. Structures and Spin Mechanism

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9. INTEGRATED ELECTRONICS

FUNCTION

The integrated electronics subsystem (Figures 9-1 and 2) is used mainly for maintenance of communications and tracking during the zero-g modes of transplanetary operations and for transmittal via downlink of scientific data during the planetary encounter operations. It is also used for performance on on-board attitude control during the eight spin modes of approximately 14 days each. During these times, the on-board G&N optical equipment will be utilized to conduct star sightings and, in conjunction with the display keyboard, the updating of the on-board computer(s). During this time, the inertial measuring unit (IMU) is aligned with respect to a celestially fixed coordinate system. Information concerning the attitude of the spacecraft is required prior to coarse alignment, since the platform alignment is accomplished by relating the platform gimbal angles to the spacecraft axes. The G&N optics will be utilized for target acquisition. After performing sightings on stars with known positions, the mark button actuation stores the optics trunnion angle, IMU gimbal angles, and time in the computer. Ephemeris data are provided in the map and data viewer. The computer commands the rotor of the coupling display unit resolver to the desired gimbal angle; the stator of the resolver is actuated by a signal representing the actual gimbal angle. The output of the resolver is an error signal that represents the difference between the actual and desired gimbal servo loop voltage to torque the platform gimbal until the actual and desired angles correspond. A fine alignment is used to refine the position of the IMU gimbals; the gimbal positions are controlled by torquing the gyros and using the resultant gyro pick-off error signal and the DC torque motors to align the gimbals. The outputs of the IMU resolvers are transmitted to the coupling display unit resolvers, and an error signal, representing the difference between the IMU and CDU resolver positions, is generated; this signal is transmitted through a 2-speed switch to the CDU servo amplifier which drives the CDU shaft motor until the CDU resolver output is nulled. The total number of pulses transmitted by the shaft encoders to the computer and the CDU display panel indicates the gimbal position for crew monitoring. Continuous tracking and ranging by MSFN tracking stations will constitute the primary mode of obtaining trajectory information. This will be accomplished by means of the on-board communications subsystem. A subsidiary function will be the transmittal of scientific and mission status data by means of the PCM telemetry equipment. The premodulation processor provides the interface connection between the spacecraft data gathering equipment and the rf electronics equipment. It

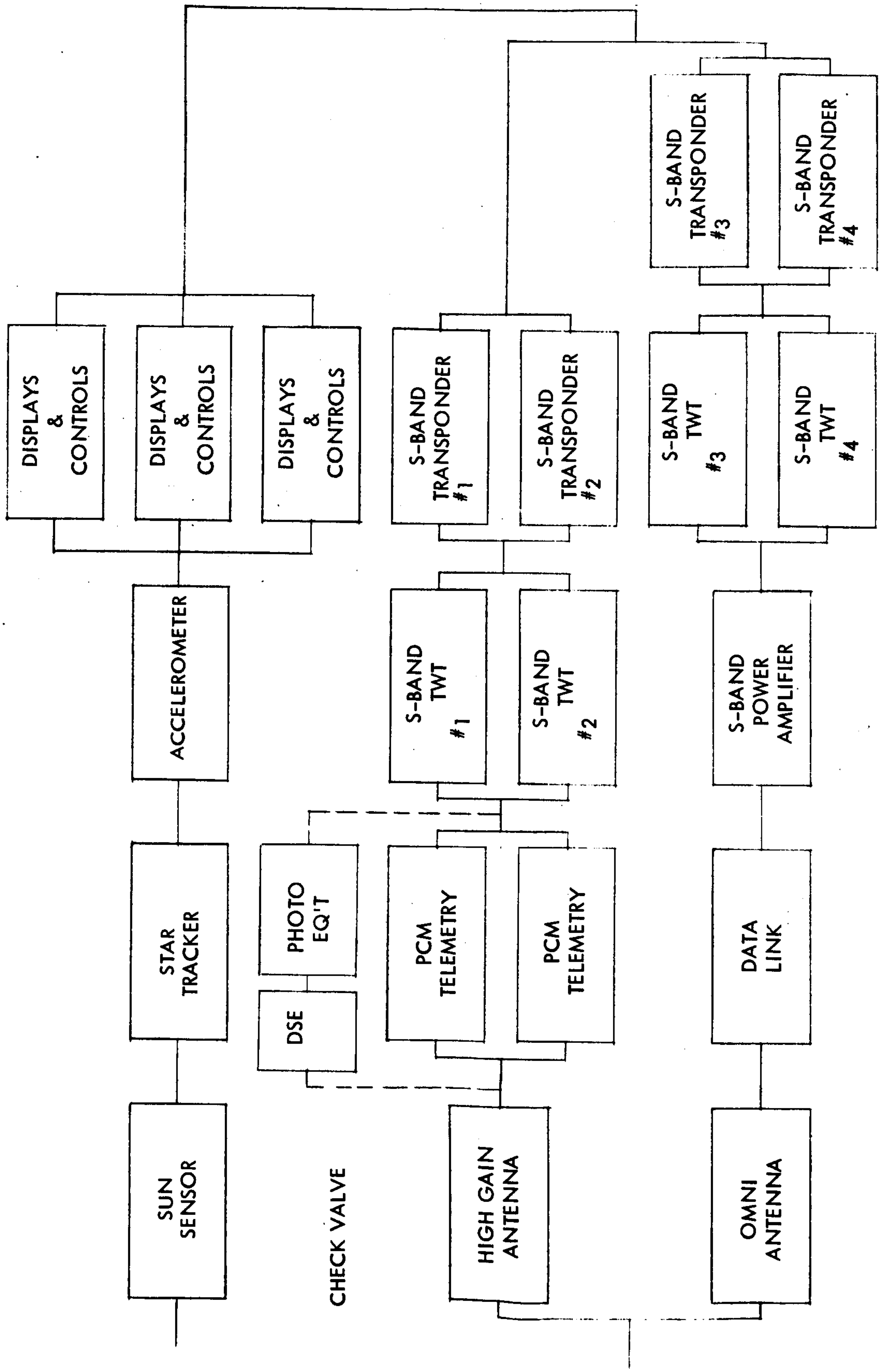

Figure 9-1. Integrated Electronics Subsystem Communications and Data Equipment

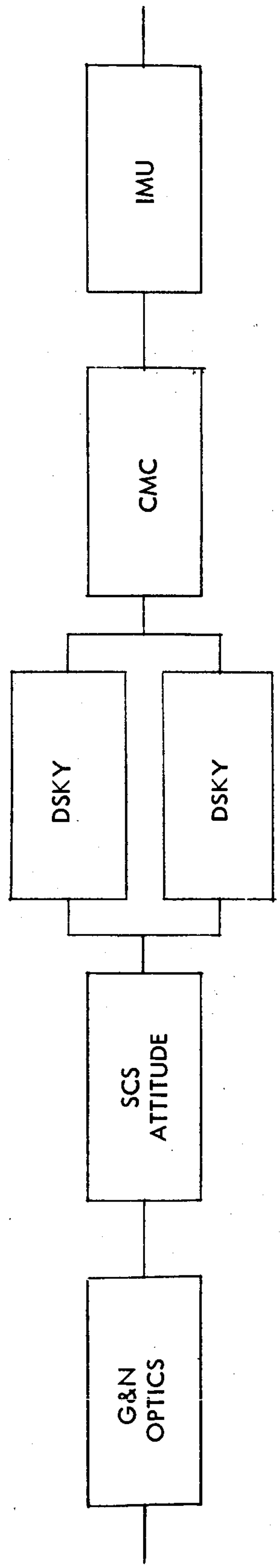

Figure 9-2. Integrated Electronic System Guidance and Control Equipment

accomplishes signal modulation, demodulation, signal mixing, and the proper switching of signals so that the correct intelligence corresponding to a given mode of operation is transmitted. Present design data indicates quadruple redundancy of the S-Band transponder equipment. In one mode, the data are transmitted via the high gain antenna during the 0 gravity mission phases; in another mode, the use of the omniantenna requires the power amplifier to be utilized in order to obtain a sufficient signal to noise ratio at the Earth receiving stations. Spacecraft orientation is maintained via the star tracker and sun sensor equipment in conjunction with the accelerometer. The S-Band equipment is a narrow-band, double conversion, superheterodyne, automatic-phase tracking receiver and transmitter operating in the 2 KMC region. Tracking is accomplished by a phase-locked loop. A stable carrier signal is transmitted to the spacecraft, filtered, frequency multiplied, and transmitted to MSFN. The ground station utilizes an automatic phase control receiver to track the phase of the spacecraft carrier by means of a reference signal. Ranging is accomplished by transmission of pseudo-random pulse trains to the spacecraft, where the spacecraft carrier is modulated for transmission back to the ground station. The time delay observed is then used to determine spacecraft range. The transmitter may operate in one of two modes—a coherent mode, in which the radio frequency signal from the receiver is fed through a phase modulator to the output, and a wide-band FM mode in which a voltage controlled oscillator is modulated by a composite waveform of telemetry and voice. The power amplifier boosts the signal to a level of either 5 or 20 watts, depending upon the cathodic potential of the TWT. The power amplifier generally achieves the same performance with the omniantenna as the high gain antenna in the coherent mode of transmission. An alternate method of retrieving scientific data during the planetary encounter is via the film cameras and tape recorder unit (Data Storage Equipment). Whether this method would constitute mission success is as yet, undetermined.

OPERATING REQUIREMENTS

Operating requirements are indicated on the time line in Section 2.

CRITICAL FUNCTIONS AND COMPONENTS

Mission simulation studies on the Recomp II computer indicate an overall electronics subsystem reliability of 0.3019 for the phases from transplanetary injection to Earth entry. Of the total mission failures, 17 percent were caused by failures in the communications equipment and 83 percent by failure in the Guidance and Control equipment. Of the former equipment, the major part (i.e., 88 percent of the 17 percent) contribution to the unreliability was due to failures in the star tracker, used for spacecraft alignment for proper orientation of the high gain antenna. Of the latter equipment, the major contributors to the 83 percent unreliability component, in descending order, were the G&N optics (46 percent), the SCS attitude electronics (32 percent), and CMC (21 percent). In addition, reliability is compromised by limited life items such as the inertial measurement unit and the travelling wave tubes, whose lives are less than that of the corresponding mission operating times. These problem areas may be alleviated partially by new techniques such as, for example, nuclear magnetic resonance-based gyros, and varactor frequency multiplication to obtain all solid state S-Band equipment. In addition, the relatively long mission time of the Mars Flyby Mission will allow sparing of such items as the Stabilization and Control attitude electronics at the subassembly, or even possibly, part level. Thus a store of capacitors, transistors, diodes, etc., with appropriate test equipment for fault isolation may be carried on-board to effect in-flight repairs. In this way, it is anticipated that the reliability objectives of the integrated electronics subsystem may be met.

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10. COMPUTER METHODOLOGY

The results of individual failure mode analyses of the subsystems of the Mars Flyby Vehicle System were encoded, and the model was entered in the Recomp II Digital Computer, which was programmed with the Maximum Entropy Simulation Program (SIMAX). This program contains two integral random number generators coded by the failure rate data for each of the individual component items of the various subsystems. This coding operation results in the simulation of operational mode failures during any one of ten pre-assigned phases of the mission, from transplanetary injection to Earth encounter, in accordance with the actual probabilities of such failures under the anticipated conditions of stress, duty cycles, etc. The primary mode of operation, which is also the highest performance mode, is analytically solved for, and a secondary random-number generator is utilized to simulate single and multiple failures of the primary mode of operation in accordance with the equipment failure rates of the items comprising the primary modes. Following this, the secondary operational modes are tested to determine the feasibility of completing the mission under the conditions of failures in both the primary and secondary operational modes. Thus, each mission simulation does not begin unless the primary mode of operation is inoperative. This procedure minimizes the length of the computer runs, since the results of the mission simulations are combined by the computer with the primary mode reliability (solved for analytically) to arrive at the final value for the mission reliability of each subsystem. The procedure also takes advantage of the quick turn-around time on the Recomp II computer. This hybrid technique (i. e., probabilistic and closed-form analytic) results in a relatively high machine efficiency comparable to accelerated testing at hyperenvironmental levels to obtain statistically significant data in short test periods. This methodology is generally described as maximum entropy, in accordance with theoretical considerations derived from information theory.

11. CONCLUSIONS AND RECOMMENDATIONS

The following pages present general reliability considerations, problem areas, and subjects which must be investigated before extremely long manned missions take place. The areas of investigation listed do not necessarily fall within the responsibility sphere of System Effectiveness. Study areas which do fall within the scope of the next study phase include: determination of feasible methods, including redesign; extension of component life; incorporation of redundancy; and integration of man's maintenance capabilities into the system.

RELIABILITY PRINCIPLES

1. Practical Design Margins

   Normal Earth operation to ASME standards approaching a 5-to-1 design margin.

   MSFC prefers margins near 2-to-1 for space vehicle structures.

   S-II experience has been borderline at 1. 5-to-1 for large cryogenic tanks.

   Requires great detail in uncovering all environment stress and material strength variations.

2. Minimum Single Point Failures

   Define single point failures as crew loss resulting from a single component failure.

   Initial design eliminates all single point failures.

   Evaluate alternate, feasible methods minimizing risk from necessary single point failures.

   Provide internal redundancy and/or large-design margins for residual single point failures.

3. Maintenance Provisions

Detection

Current measurements are insufficient to detect all critical failures.

Automatic versus manual monitoring has not been defined.

Diagnosis

Complicated equipment failures are extremely difficult to diagnose for causes.

Earth-based computer diagnosis must be available to assist.

Correction

Adequate spares or repair capability must be provided.

Astronaut training must include time studies of possible replacements.

4. Demonstration Beyond Maximum Environments

Overstress tests will determine actual design margins.

Extrapolate parameter drift tests to indicate long duration performance.

Design number and kind of experiments for maximum information.

5. Resolution of all Failures

Cause and correction of each failure must be a major effort.

Individual component reliability demonstration only after a failure.

GENERAL PROBLEM AREAS

1. Materials

Specialized materials are being required to withstand novel environments for longer than specified in any current tests. The natural environments are new, and the induced environments require special consideration. Sealing materials are particularly susceptible because of the successive deterioration with proceeding mission phases due to creep, chemical attack, radiation breakdown, and vacuum welding.

2. Longevity

Many components in current spacecraft will wear out or have insufficient MTBF with increasing mission duration. They will require repair or replacement. The increase in spares requires careful analysis and trade-off among weight, design margin, and redundancy criteria.

3. Contamination

The ability to remove all significant contamination before flight has not been demonstrated on current spacecraft. Small amounts of contamination in critical areas could jeopardize mission success at any time. Metal to metal seats are particularly susceptible, and cabin atmosphere cleanliness is critical.

4. Effects of Multiple Environmnets

Single environments are severe. The effects of multiple environments are unknown. Addition of stresses may increase the load requirements beyond the present earthbound capability for testing.

5. New Requirements

In general, new designs require a development period in which the performance and reliability capability increases; e. g. , the artificial gravity-spin system will require careful design and test before being available for transplanetary flight.

6. Crew-Machine Task Mix

The new requirements of a long, independent mission must be identified in terms of an optimum mix of human and machine tasks using the best features of each.

7. Optimization for Multiple Missions

In current programs optimization adjustments have not been performed for various reasons. However, recognition of multiple objectives, the emphasis to be placed on each, the mathematical model of the system, including incentives, and the optimization by typical dynamic programming methods are essential to the efficient conduct of any large future program.

8. Unknown Environments

The first few missions will be critical in their encounter of all actual environments. The variations of present knowledge of space characteristics are too wide to allow a definitive design; e.g., meteoroid flux level uncertainty changes solar panel size by an order of magnitude.

SPECIFIC INVESTIGATIONS

1. Space Environments on External Seals

Relief valves - vacuum welding of seats on long exposure.

Hatches - flow of current soft material under long-term compression.

Gimbal joints - vacuum welding of joints on long exposure, or high reliability bellows.

Engine start valves - Flow of current soft seat in SPS valve for long passive exposure.

2. Retention of Fluids for Extended Duration

Attitude control and spin motor propellants - the present positive expulsion bladder has not been tested for the intended duration, but indication is that excessive gas leakage could occur.

Main engine propellants - the large weight requirements make redundancy expensive, and all current vehicles rely on excess margin to compensate for leaks. For this single source, one meteoroid puncture would prevent midcourse correction.

Seal compatibility for extended duration - no known soft sealing material is compatible with present corrosive propellants (for the durations required).

Helium permeation through tank materials - intermolecular seepage of helium through structural materials may prevent retention of helium in high pressure tanks for a mission of long duration.

Creep of materials under stress - titanium tanks are notorious for creep and strength reduction with time.

3. Meteoroid Frequency, Prevention, and Repair

Evaluation of risk of encounter - current knowledge is not accurate enough to use in subsystem trade-off studies; e.g., solar panel life.

Structure and materials for energy dissipation - new materials and configurations will be required to prevent puncture of critical areas while remaining light in weight.

Detection of penetration - passive components and inaccessible locations may prevent detection of failures due to penetrations.

Emergency procedures - detailed study of emergency procedures must account for detection, analysis, and correction of failures due to penetration.

4. Main Engine Critical Components

Outgassing of ablative material - rocket engine chamber cooling ablative material is a plastic impregnated glass cloth which changes characteristics on long exposure to vacuum.

Single versus multiple engines - although much internal redundancy reduces criticality to a minimum, the uncertainties of rocket combustion make dependence on a single engine for successful return a high risk. Multiple engines would reduce that risk but require redesign.

Gimbal method - hard links, bearing joints, and uncertain responses make current gimballing methods critical to success, since complete redundancy is difficult to achieve. Gas ingestion or flexible joints could improve reliability by complete redundancy and simplicity.

5. Artificial Gravity Methods and Ramifications

Extension and rotation initiation - new equipment and methods are required.

Stability during rotation - since the masses are not point-concentrated, and movement within each is required, deviations in rotation plane and wobble need to be overcome without excessive expenditure of propellants or damper weight.

Communication during rotation - communication between Earth and rotating parts is difficult without power enough for an omni transmission. The requirement of continuous antenna orientation throughout rotation would be an added reliability, weight, and control problem.

Guidance during rotation - star sighting would be difficult without the added complexity of continuously pointable optics.

Retraction and rotation cessation - careful control during de-spin will be required to dissipate energy and dock without damage.

6. Reentry Environments

Acquisition of reentry corridor - man's ability to guide the vehicle to the corridor should be determined.

Outgassing of ablative material - the ablative material is a plastic impregnated material which changes characteristics with vacuum exposure time.

Deterioration of parachute material - vacuum and radiation cause strength loss.

Deterioration of parachute motors - exposure of solid propellant to vacuum, radiation, and temperature variations causes changes in burning characteristics.

7. Inflight Failure Processing

Detection of relay drift - drift in relay setting could cause inadvertent operations and inability to operate; e.g., no automatic recharging of batteries.

Detection of residual propellant error - propellant management is critical, so measurement of residual propellant must be accurate enough for timely warning of over-use.

Failure signal hierarchy - studies must be made to trade off automatic and manual failure detection means.

Computer programs for diagnosis - computer programs must be available to the astronaut for rapid failure diagnosis; e.g., on-board or Earth-based equipment.

Resetting and recalibration techniques - the requirement of maintenance means checking and recalibration of gages and equipment which drift due to longevity and space environments.

8. Wearout of Rotating Machinery

Glycol pumps in ECS circuit - insufficient operating life on current pumps.

Combined rotating unit in EPS - current life is insufficient; requires development.

Gyros in guidance and stabilization systems - the mission duration precludes accuracy, unless shutdown during spin is allowed.

9. Battery Integration into EPS

Reentry battery recharging - current batteries will lose charge without load over a long period; e. g., six months. Reentry battery recharging must be planned and tested.

Pyro battery recharging - a rearrangement of method must be made to incorporate pyro batteries into the overall electrical loop for recharging; i. e., completely separate circuits are not feasible.

10. Man's Ability to Control Vehicle with Simple Aids

Manual determination - need and initiation of impulse vector

> Midcourse correction - critical and necessary functions with accuracy limits.
>
> Planetary turn - critical and necessary functions with accuracy limits.
>
> Reentry corridor - critical and necessary functions with accuracy limits.

Minimum tools and skills required - astronaut abilities and tools needed to supplement Earth guidance.

Optimum man-machine task mix - study of simple manual tools to replace complex automatic equipment requires consideration of capabilities of each in the mission context.

11. Environmental Control

Heating and cooling of propellants - main propellant freezing in exposed areas near the main valves and gimbal piping requires continuous heating, which causes a continuous power drain and the possibility of unreliable heaters. Other methods should be studied.

Insulation of cryogenic oxygen for extended duration - long manned missions will require large quantities of oxygen. Efficient storage is difficult since current insulations are insufficient to prevent excessive heat transfer.

Small rocket engine heating - the small rocket chambers need heating to prevent explosion when started. Since the heaters cause power drain, alternate heating methods should be studied.

12. Biomedical Requirements

Psychology of long confinement - hazardous, monogamous confinement in restricted area may prevent efficient performance of manual tasks.

Surgical tools and processes - long abort times require consideration of all life saving functions in-flight; e.g., surgery.

Waste processing - water and oxygen reclamation is critical to a vehicle of feasible size.

APPENDIX A. FUNCTIONAL ANALYSIS

This appendix is devoted to a functional description of the baseline Mars/Venus Flyby Mission. The seven first-level phases from 3.6 (Perform Transplanetary Injection) through 3.12 (Perform Earth Retro) are emphasized because they constitute the challenge to reliability of long storage, long duration operation, space environments, assembly, and maintenance capability. Each first level phase is further divided into several second-level functions. Block diagrams (AFSC 375-5) are provided to illustrate functional interrelationships. The necessity of each second level function to crew safety and mission success has been assessed, and reliability logic diagrams have been constructed. Three categories of necessity were used:

1. Crew Safety Critical - Failure of the function would cause crew loss.
2. Mission Success Required - Failure of the function would cause total or partial loss of mission objectives.
3. Alternate - A failed function would be compensated for by another function.

The following crew safety critical functions will be considered carefully in any subsequent hardware evaluation:

1. Inability to escape from booster explosion.
2. Inability to compensate for booster thrust misalignment.
3. Inability to jettison booster.
4. Loss of all or part of environment and life support functional control, of oxygen concentration, temperature level, pressure level, and atmospheric contamination.
5. Inability to orient spacecraft.
6. Inability to provide trajectory connections from thrust misalignment or loss.
7. Failure to de-spin.

8. Loss of individual environment control during EVA.

9. Misapplication of retro thrust.

PHASE 3.6 PERFORM TRANSPLANETARY INJECTION

3.6.1 Perform Navigational Operations - Completely redundant with the ground computer/tracking operation.

3.6.1.1 Perform Stability and Control Functions - Stability control is a prerequisite for transplanetary injection orientation and reentry. Much internal redundancy is possible.

3.6.2 Perform Guidance Functions for Injection - Peculiar to spacecraft operations, e.g., pointing spacecraft in right direction.

3.6.3 Perform Countdown Operations for Injection - Although there is a launch window, it should not be so tight that the mission would be aborted at any failure. This must be in parallel with the possibility of repair of any failure in time to enter the launch window.

3.6.4 Perform Crew and System Status Monitoring - Crew safety is not affected by the monitoring instrumentation; mission success is affected by aborts when instruments give a poor reading. Policy should be to use complementary instrumentation for all critical functions and perhaps all mission success functions as well.

Measurements anticipatory of a critical item failure could cause crew loss on failure to detect a failure, but this constitutes at least two failures; and with complementary instrumentation, more than two. This function is in parallel with the correct functioning of the equipment.

3.6.4.1 Completion of Status Check - Mission success items, not crew safety. Must be performed before proceeding.

3.6.5 Apply Earth Control Guidance and Navigation Correction Data - Redundant with data generation within the spacecraft.

3.6.6 Initiate Injection Propulsion (Booster) - Complicated process controlled entirely within the spacecraft. Critical failure modes are explosion and severe thrust misalignment.

3.6.7 Perform Injection Control and Monitoring Functions - These monitoring functions are in parallel with correct equipment operation. During firing, the procedure should be sufficiently automatic that personal control need only be exercised when failure has occurred. Critical failure modes are explosion and severe thrust misalignment.

3.6.7.1 Decision to Proceed - The spacecraft captain must look at all operations and consciously decide to go on. This is the last opportunity to return to Earth in a short time abort.

3.6.8 Terminate Injection Propulsion (Booster) - Automatic on achievement of sufficient ΔV. This could be a difficult operation because of thrust variations on multiple boosters or the necessity to cool the engine after main thrust is over. Critical failure modes: (1) explosion, (2) severe thrust misalignment, (3) expulsion of all H_2 before cooling reactor.

3.6.9 Provide I. U. Trajectory Information from Booster - This could be extensive or minimal depending on spacecraft philosophy. If boosters are standard modules, they may not contain any guidance equipment; or special gyros may be included to save time on the spacecraft guidance gyros.

3.6.9.1 Jettison Booster - Critical function. Send to solar orbit.

3.6.10 Provide Earth/Spacecraft Communications - Desirable but unnecessary. Communication is desirable (1) to assist in failure diagnosis, (2) for psychological extension of the astronaut's mental environment, (3) for transmission of experiment results, and (4) for backup guidance data, etc.

3.6.11 Provide Environment and Life Support Functions for Crew - Obviously critical to maintain crew functioning and, ultimately, life.

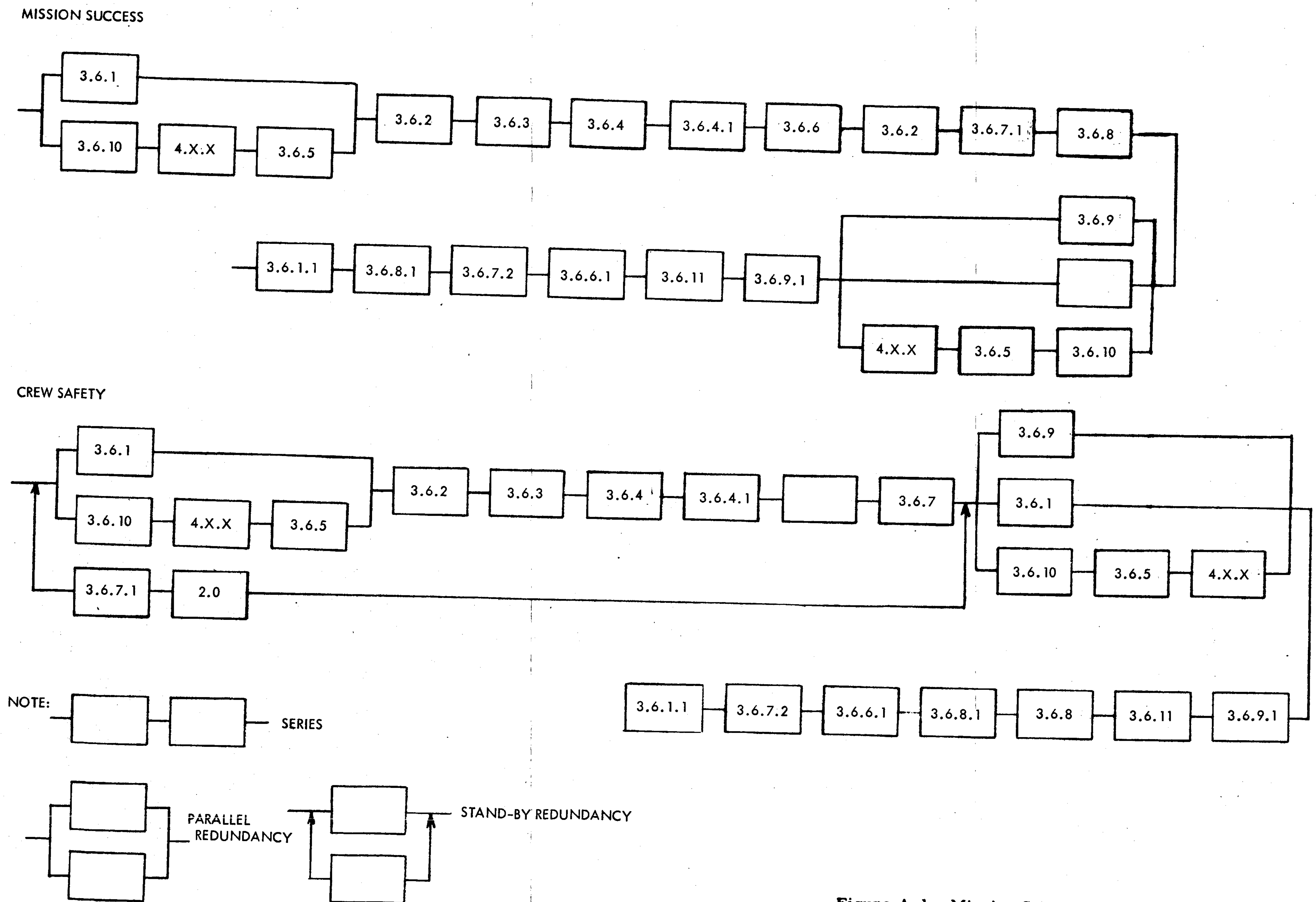
<u>Functions That Apply to More Than One Phase</u>

4. Perform MSFM Ground Tracking and Control Function - Again, desirable as backup for several functions, but not critical in itself.

2.0 Perform Abort Operations (As Required) - On a standby basis after a critical failure.

Series Functions	
Mission Success	Crew Safety
3.6 Perform Transplanetary Injection Functions 3.6.2 Perform Guidance Functions Peculiar to Spacecraft 3.6.3 Perform Countdown Operations. (Including Holds, Repair, and Restart)	

Series Functions (Cont)	
Mission Success	Crew Safety
3.6.4 Critical Measurement Success	
3.6.4.1 Completion of Status Check. (Including Command Decisions to Proceed Without a Full Check.)	
3.6.6 Initiate Injection Propulsion	
3.6.6.1 Critical Injection Propulsion Operational Modes	3.6.6.1 For explosion failure. For severe thrust misalignment.
3.6.7 Injection Propulsion. (Including possible human actions to repair or compensate for failures.)	
3.6.7.1 Decision to Proceed	
3.6.7.2 Critical Injection Propulsion Operational Modes.	3.6.7.2 For explosion failure. For severe thrust misalignment.
3.6.8 Terminate Injection Propulsion	
3.6.8.1 Critical Injection Propulsion Operational Modes	3.6.8.1 For explosion failure. For severe thrust misalignment. Continuing until all H_2 is exhausted (no reactor cooling).
3.6.9.1 Jettison Booster	3.6.9.1 Depends on design; e.g., interferes with midcourse connection thrust, overheats after shutdown (nuclear incident), etc.
3.6.11 Provide Environment and Life Support	3.6.11 No Oxygen No Temperature Control No Pressure Control No Ridding of Gaseous Diluents
3.6.1.1 Perform Stability and Control Functions	3.6.1.1 Inability to orient for reentry retro.



FOLDOUT FRAME 1

Figure A-1. Mission Success (1)

FOLDOUT FRAME 2

PHASE 3.7 PERFORM TRANSPLANETARY LEG OPERATIONS

3.7.1 Determine Trajectory Parameters - Spacecraft guidance for early correction is redundant with ground tracking and communication.

3.7.2 Compare Trajectory Data with Earth Computed Data - Backup operation.

3.7.3 Apply ΔV Correction - Once commenced, it is important to apply correct amount and direction; but corrections can be made after an error. In fact, early correction may not be necessary; it corrects for possible inaccuracy in injection guidance. Criticality depends on spacecraft configuration. Is there enough propellant to correct later on for a missed early correction? Can the ΔV be applied at any time during an extended time period? Methods should be developed which continually compute the direction and duration of thrust so that any hold in correction initiation will not jeopardize the mission, and it is important to hold initiation until all factors are correct since, within limits, it should be less degrading to wait a short time than to make two corrections. Corrections should be made only when guidance shows a clear requirement, not within the error cone (of specified accuracy, e.g., 3σ). Theoretically, continuous correction would require less propellant expenditure if corrections were always made in the right direction; but this condition cannot be assured. Corollary corrections should increase in frequency nearer the target since errors become more easily detectable.

3.7.4 Recalculate Trajectory Parameters - Spacecraft guidance for mid-course correction should still be redundant with ground tracking and communication.

3.7.5 Completion of a Complete Spacecraft Checkout - For crew safety, checkout is always in parallel with the successful operation of the equipment. In this case, completion of the checkout is not critical for mission success. It is simply desirable to know the status. Changing the place of checkout so that it follows artificial gravity maneuvers was done because (1) no abort is possible if a problem exists; (2) a checkout is scheduled only a short time before; and (3) entering artificial gravity as soon as possible is desirable since the astronauts have more time to make a complete check.

3.7.6 Prepare for the Artificial Gravity Mode - Only spacecraft personnel can do this, but at a non-emergency pace. The possibility that a hold can be made for a repair is good.

3.7.7 Extend Spacecraft, Achieve One-Third Gravity, and Stabilize - This is a most difficult maneuver when two masses are on the ends of a string (bolas). The maneuver is clearly necessary for mission success since excessive debilitation of astronauts is likely if zero g is maintained throughout the flight. However, crew safety—getting back alive—may not be affected; i. e. , the return trajectory and propulsion can be controlled from the ground, and the astronauts will be required to perform extra physical exercises. It may be desirable to provide an alternate extension means for redundancy.

3. 7. 8 Transplanetary Coast (First Phase) - Used to denote passage of time only. When specific activities are planned, they will be inserted.

3. 7. 9 Accept and Apply Earth Computed Trajectory Data - Backup operation to spacecraft guidance.

3. 7. 10 Determine the Need for Trajectory Correction - Decision is made in spacecraft on basis of ground and spacecraft data. However, since the ground crew is capable of making this decision also, internal redundancy exists.

3. 7. 11 Retract Spacecraft and Return to Zero g Mode - This is a most difficult and critical maneuver, vital to mission success. When in the artificial gravity mode, it is necessary for crew safety. Abort possibilities exist, depending on the configuration, e. g. , mid-course correction propulsion and the reentry capsule escape from the whirling system.

3. 7. 12 Apply ΔV for Mid-Course Correction - See 3. 7. 3.

3. 7. 13 Recalculate Trajectory Parameters - See 3. 7. 4.

3. 7. 14 Same as 3. 7. 6.

3. 7. 15 Same as 3. 7. 7.

3. 7. 16 Same as 3. 7. 8 for second phase.

3. 7. 17 Same as 3. 7. 9.

3. 7. 18 Perform Interplanetary Experiment Program - Each experiment can be considered a part of the mission success, but much time can be spent setting them up and repairing when necessary.

Functions that Apply to More Than One Phase

3. 7. 1. 1 Perform Stability and Control Functions - Every zero g mode must activate the stability control system (RCS), which is inactive during the zero g mode. The RCS may even be called on to start and stop the rotation for artificial gravity. A lot of redundancy can be justified since this system is critical for mission success and crew safety.

4.X.X. See 3.6.

3.6.4 See 3.6. Vital to mission success only when an abort mode is available.

3.6.10 See 3.6.

3.6.11 See 3.6.

Series Functions	
Mission Success	**Crew Safety**
3.7.1.1 Perform Stability and Control Functions	3.7.1.1 Inability to orient for ΔV corrections.
3.7.3 Apply ΔV Correction	3.7.3 Inability to correct velocity and direction; the reentry window therefore missed.
3.7.6 Prepare for the Artificial Gravity Mode	
3.7.7 Extend Spacecraft, Achieve One-Third Gravity, and Stabilize	
3.7.11 Retract Spacecraft and Return to Zero g Mode	3.7.11 Continue to spin without ability to correct trajectory for reentry.
3.7.18 Perform Interplanetary Experiment Program	3.6.11 See 3.6.
3.7.8 Transplanetary Coast (First Phase)	3.7.8 The spacecraft must go through this time period successfully; however, when specific tasks are determined, they will be inserted.
3.7.16 Transplanetary Coast (Second Phase)	3.7.16 See 3.7.8.

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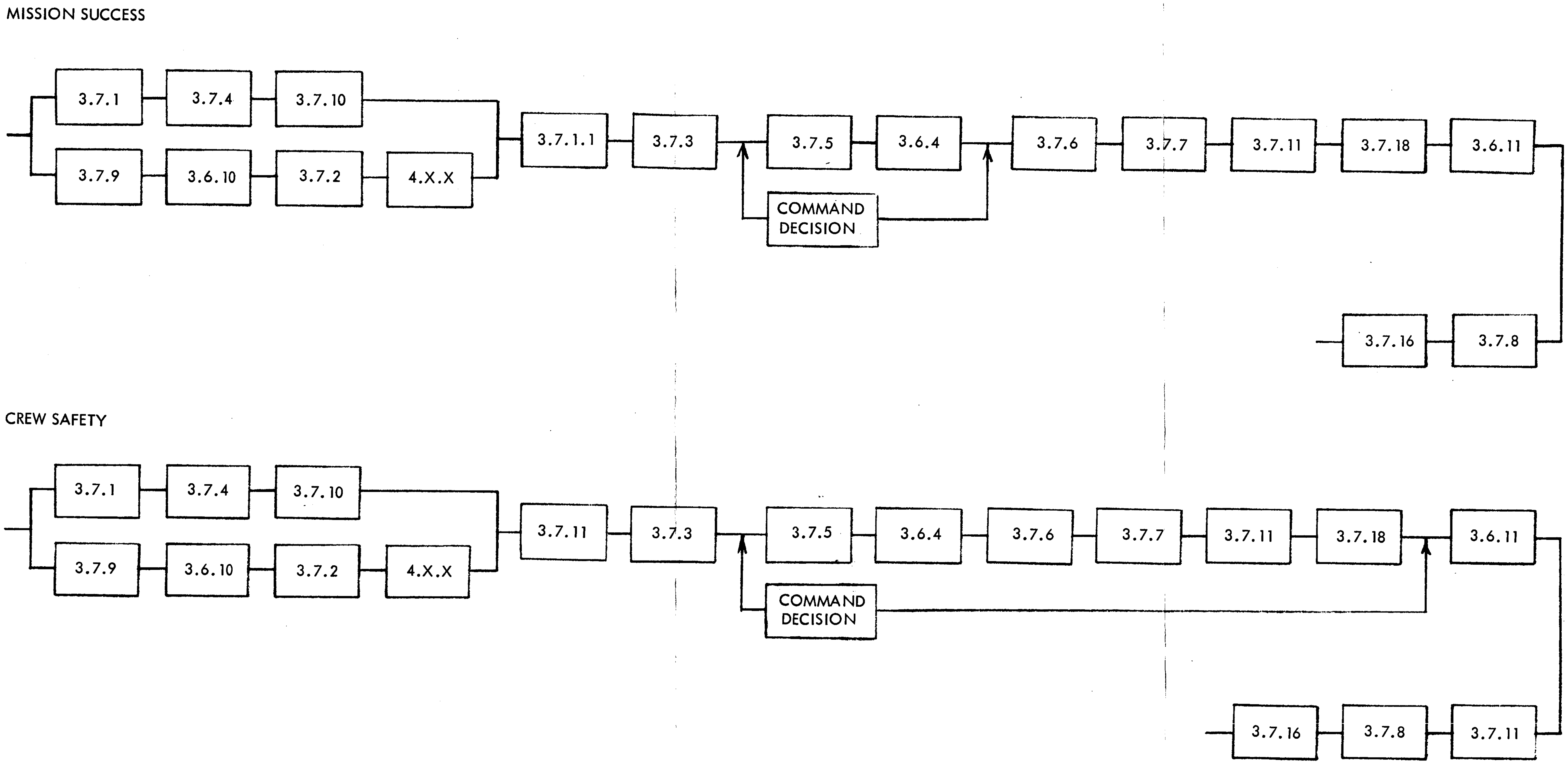


FOLDOUT FRAME 1

Figure A-2. Mission Success (2)

FOLDOUT FRAME 2

PHASE 3.8 PERFORM PLANETARY APPROACH OPERATIONS

3.8.1 Prepare to Return to Zero g Mode - This intermediate step is unnecessary since 3.7.11 included both preparation for and completion of the return to zero g.

3.8.2 Accept and Insert Earth-Computed Trajectory Data - This now is a courtesy backup, because the proximity of the planet and vast distance from earth preclude accurate guidance.

3.8.3 Return to Zero g Mode - See 3.7.6 and 3.7.7.

3.8.4 Perform Spacecraft Stability Functions - See 3.7.1.1.

3.8.5 Calculate Trajectory Change Required and ΔV - Can be considered redundant with Earth-based tracking only in a gross sense. Although still shown in parallel, for crew safety, the probability of success of the Earth tracking is very low. This can really be considered in series for mission success and redundant for crew safety.

3.8.6 Apply ΔV Correction - See 3.7.3.

3.8.7 Recalculate Trajectory - See 3.8.5.

3.8.8 Continue Planetary Approach Coast - Need be used only to designate passage of time. Should be replaced by actual functions covering the time period.

4.X.X. See 3.6.

3.6.4 See 3.6.

3.6.10 See 3.6.

3.6.11 See 3.6.

Series Functions	
Mission Success	Crew Safety
3.8.3 Return to Zero g Mode	3.8.3 See 3.7.11.
3.8.4 Perform Spacecraft Stability Functions	3.8.4 See 3.7.1.1.

Series Functions (Cont)	
Mission Success	Crew Safety
3.8.5 Calculate Trajectory Change Required and ΔV	
3.8.6 Apply ΔV Correction	3.8.6 See 3.7.3.
3.6.11 Provide Environmental and Life Support Functions	3.6.11 See 3.6.

PHASE 3.9 PERFORM PLANETARY ENCOUNTER OPERATIONS

3.9.1 Prepare Encounter Support Systems - Systems on the spacecraft which support the experiments are as follows: communication with excursion craft, tracking and guidance of excursion craft, electrical power within the spacecraft, temperature control, location brackets, and instrumentation. Ideally, these could be accomplished automatically; but practically, some adjustments, placements, and measurements will be accomplished by astronauts. All of these myriad individual items are required for complete mission success, but they can be accomplished over a relatively long time period.

3.9.2 Perform Spacecraft Attitude Control Function - See 3.7.1.1.

3.9.3 Prepare Planetary Scientific Equipment - Work on the experimental equipment itself may require adjustment, locating (inside and outside spacecraft), calibration of instruments in place, and construction of final form. The only time crew safety is involved is with possible crew loss during required EVA on suit or tether failure.

3.9.4 Initiate Optical Sensor Functions - Specialized actions for mission success.

3.9.5 Perform Lander Probe Deployment Functions - Specialized actions for mission success.

3.9.6 Initiate Photographic Sensor Functions - Specialized actions for mission success.

3.9.7 Confirm Lander Probe Trajectory - Tracking of lander probe - mission success.

3.9.8 Establish Communications with Lander Probe - Radio contact with lander probe - mission success.

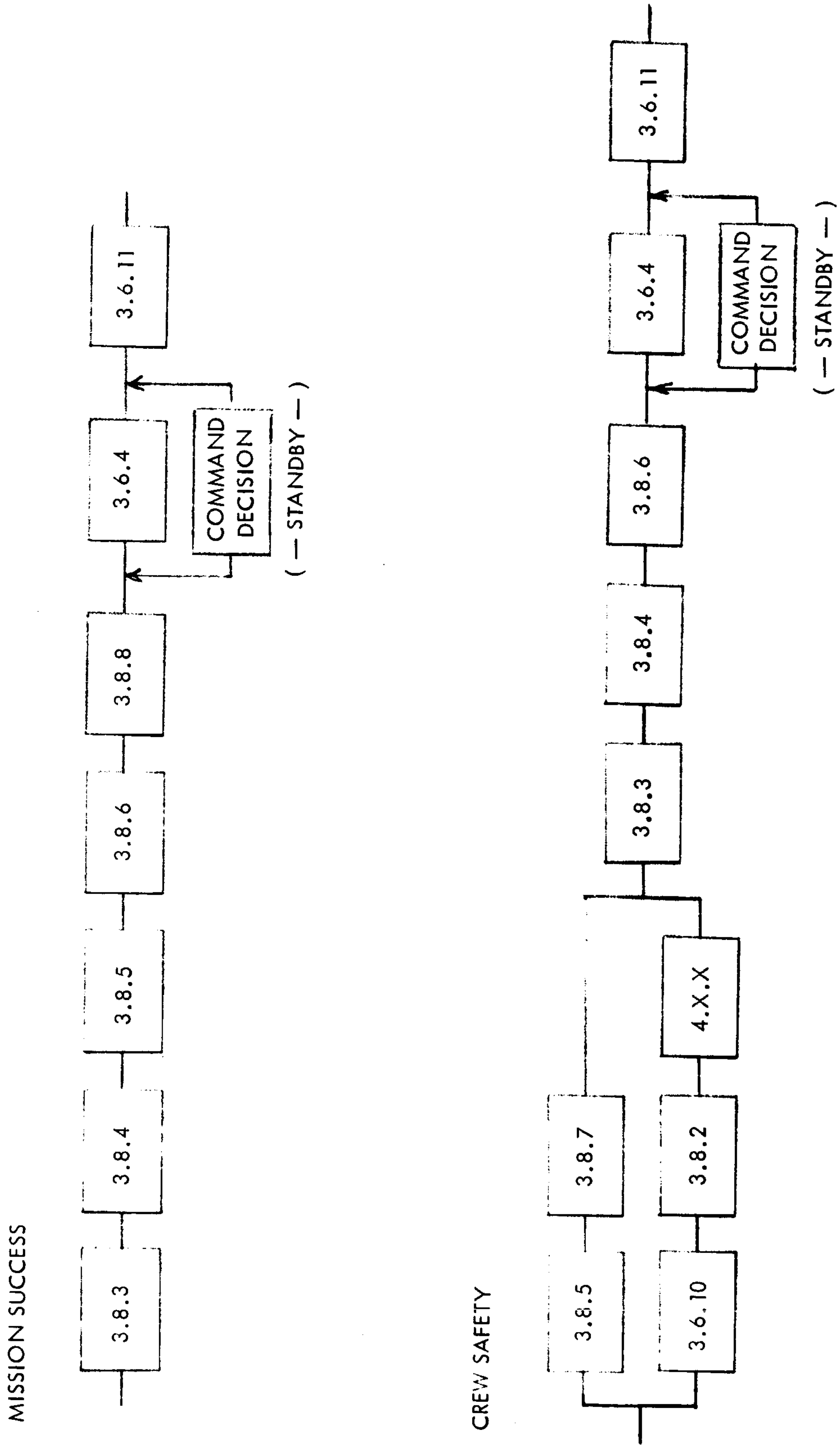


Figure A-3. Mission Success (3)

3.9.9 Perform Lander Probe Tracking and Control Functions - Specialized control functions for lander probe - mission success.

3.9.10 Perform Lander Probe Data Storage and Relay Function - Special mission success functions.

3.9.11 Perform Orbital Probe Insertion(s) Function - Special mission success actions.

3.9.12 Perform Orbital Probe Tracking and Control Functions - Special mission success actions.

3.9.12.1 Confirm Orbital Probe Trajectory - Special mission success actions.

3.9.13 Establish Orbital Probe to Spacecraft Communication - Special mission success actions.

3.9.14 Perform Orbital Probe Data Storage and Relay Function - Special mission success actions.

3.9.15 Perform Encounter Experiments - Special mission success actions.

3.9.16 Relay Data to Earth Control - Much depends on the spacecraft configuration for mission success requirements. For minimum data storage in spacecraft, relay of data may be necessary to return all experiment information. For complete mission success, relay must therefore occur. Design requirement should be more liberal with data storage space. Data relay should be used as the preferred means, data storage as an alternate, redundant means. Even if every other point is retained, sufficient data storage should be provided for a minimum mission success.

3.9.17 Terminate Planetary Encounter Experiments - Many nonautomatic, mission success actions.

3.9.17.1 Summarize Data in Transmittal Form - The voluminous data obtained from the experiments would glut available communication channels and tax the power level. The astronauts should review preliminary readout data and transmit summaries only as the first result. Later, as the spacecraft approaches Earth, more of the data can be sent at reduced power and with greater clarity.

3.9.17.2 Store Data - Sufficient storage capacity should be provided on the spacecraft for return of all required mission success information. This is in parallel with relay data.

Functions that Apply to More than One Phase

4.X.X. See 3.6. Can be omitted from this phase since no trajectory corrections are expected.

3.6.4 See 3.6

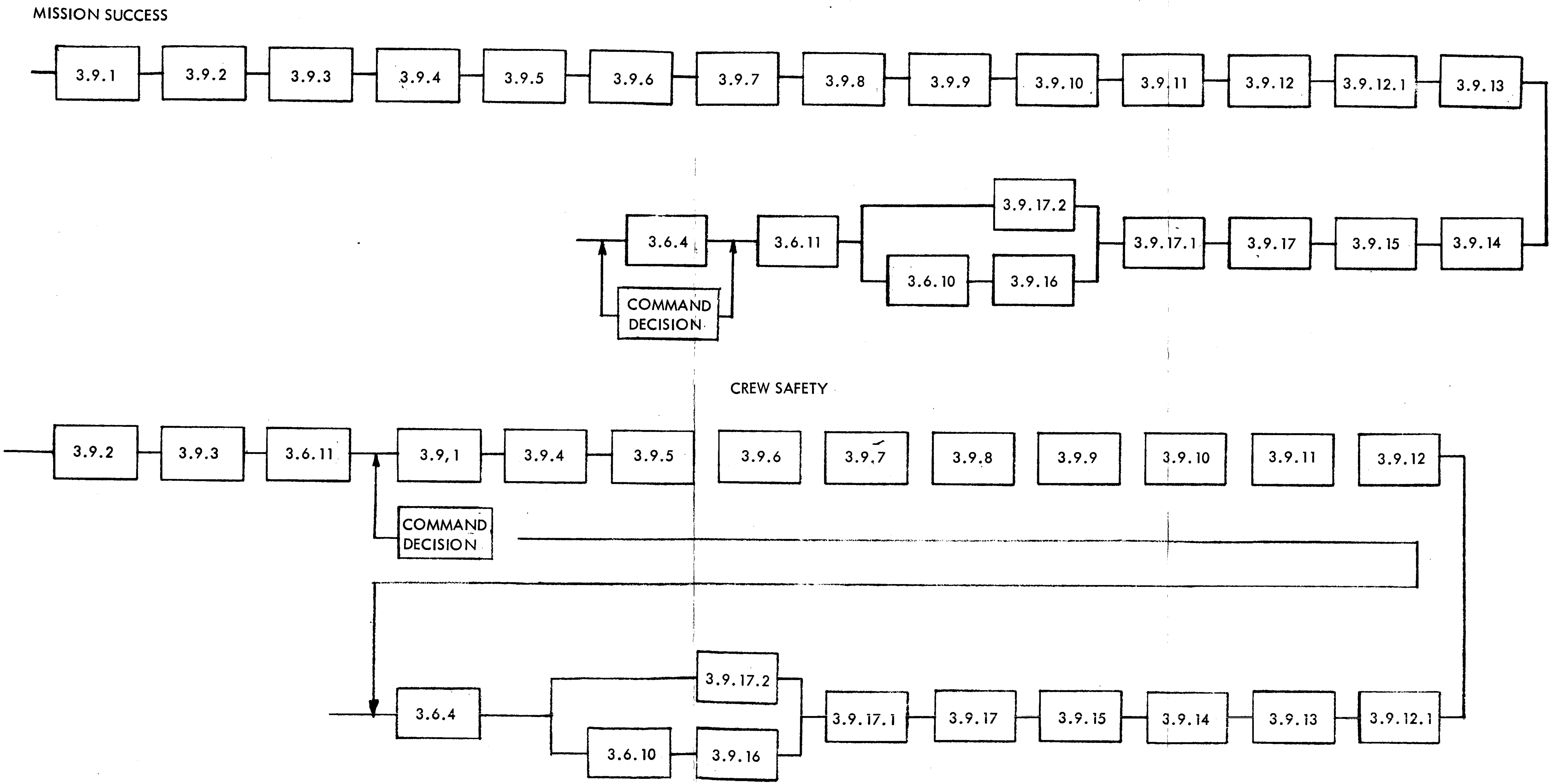
3.6.10 See 3.6.

3.6.11 See 3.6.

Series Functions	
Mission Success	Crew Safety
3.9.1 Prepare Encounter Support Systems	
3.9.2 Perform Spacecraft Attitude Control Function	3.9.2 See 3.7.1.1.
3.9.3 Prepare Planetary Scientific Equipment	3.9.3 For EVA suit and tether failure.
3.9.4 Initiate Optical Sensor Functions	
3.9.5 Perform Lander Probe Deployment Functions	
3.9.6 Initiate Photographic Sensor Functions	
3.9.7 Confirm Lander Probe Trajectory	
3.9.8 Establish Communications With Lander Probe	
3.9.9 Perform Lander Probe Tracking and Control Functions	
3.9.10 Perform Lander Probe Data Storage and Relay Function	
3.9.11 Perform Orbital Probe Insertion	
3.9.12 Perform Orbital Probe Tracking and Control Functions	
3.9.12.1 Confirm Orbital Probe Trajectory	
3.9.13 Establish Orbital Probe to Spacecraft Communication	
3.9.14 Perform Orbital Probe Data Storage and Relay Function	
3.9.15 Perform Encounter Experiments	
3.9.17 Terminate Planetary Encounter Experiments	
3.9.17.1 Summarize Data in Transmittal Form	

Functions that Apply to More than One Phase

3.6.11 Provide Environmental and Life Support Functions - See 3.6.



FOLDOUT FRAME 1

Figure A-4. Mission Success (4)

FOLDOUT FRAME 2

PHASE 3.10 PERFORM TRANSEARTH OPERATIONS

3.10.1 Determine Trajectory Parameters - See 3.8.5.

3.10.2 Compare Trajectory Data with Earth Computer Data - See 3.8.2.

3.10.3 Apply ΔV Correction - See 3.7.3.

3.10.4 Recalculate Trajectory Parameters - See 3.10.1.

3.10.5 Perform a Special Spacecraft Status Check - After all the planetary encounter experiments are completed and the spacecraft is in a state of artificial gravity, a complete check will apprise the astronauts of the condition of the spacecraft. This is the longest portion of the trip, and the craft must be maintained carefully. The check is still in parallel with successful operation of the equipment and can be made over a long period.

3.10.6 Prepare for Artificial Gravity Mode - See 3.7.6.

3.10.7 Extend Spacecraft, Achieve One-Third g, and Stabilize - See 3.7.7.

3.10.8 Transearth Coast (First Phase) - Passage of time only. Need specific functions during this time period.

3.10.9 Perform Transearth Experiment Program - Each experiment is mission success.

3.10.10 Accept and Apply Earth Computed Trajectory Data - See 3.7.9.

3.10.11 Determine Need for Trajectory Correction - See 3.7.10.

3.10.12 Retract Spacecraft and Return to Zero g Mode - See 3.7.11.

3.10.13 Apply ΔV for Midcourse Correction - See 3.7.3.

3.10.14 Recalculate Trajectory Parameters - See 3.7.4.

3.10.15 Same as 3.10.6.

3.10.16 Same as 3.10.7.

3.10.17 Transearth Coast (Second Phase) - See 3.10.8.

3.10.18 Same as 3.10.10.

3.10.19 Initiate Earth Approach Experiment Program - Another set of mission success actions.

Functions that Apply to More than One Phase

3.9.2 Maintain Spacecraft Attitude Control Function - See 3.7.1.1.

4.X.X. See 3.6.

3.6.4 See 3.6.

3.6.10 See 3.6.

3.6.11 See 3.6.

Series Functions	
Mission Success	Crew Safety
3.10.3 Apply ΔV Correction	3.10.3 See 3.7.3.
3.10.6 Prepare for Artificial Gravity Mode	
3.10.7 Extend Spacecraft, Achieve One-Third g, and Stabilize	
3.10.8 Transearth Coast (First Phase)	3.10.8 Depends on what critical equipment is provided for this phase of the mission.
3.10.9 Perform Transearth Experiment Program	
3.10.12 Retract Spacecraft and Return to Zero g Mode	3.10.12 See 3.7.11.
3.10.13 Apply ΔV for Midcourse Correction	3.10.13 See 3.7.3.
3.10.15 Same as 3.10.6	
3.10.16 Same as 3.10.7	
3.10.17 Transearth Coast (Second Phase)	
3.10.19 Initiate Earth Approach Experiment Program	
Functions that Apply to More than One Phase	
3.9.2 Maintain Spacecraft Attitude Control Function	3.9.2 See 3.7.1.1.
3.6.11 Provide Environment and Life Support Functions for Crew	3.6.11 See 3.6.11.

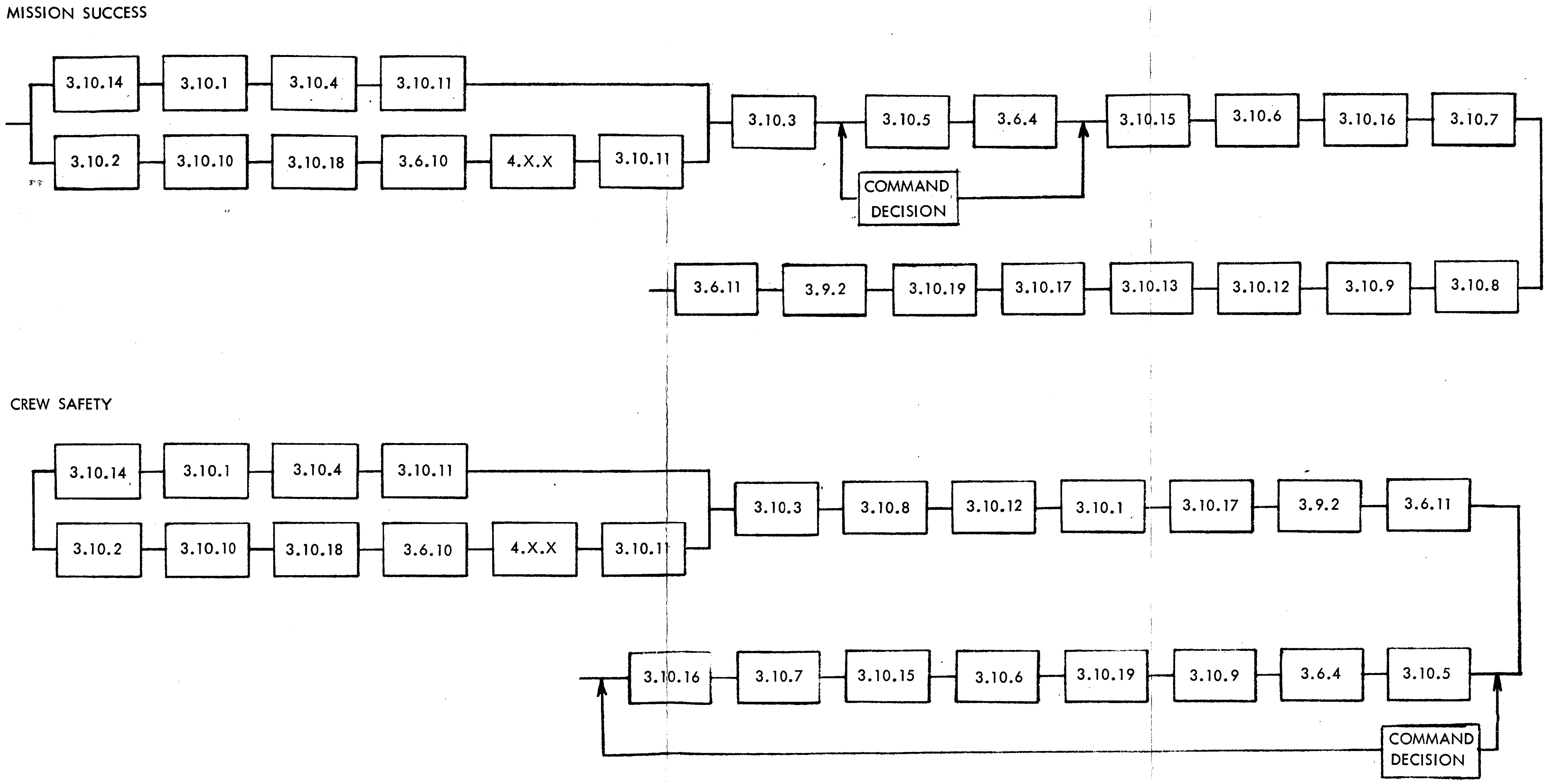
PHASE 3.11 PERFORM EARTH APPROACH OPERATIONS

3.11.1 Prepare to Return to Zero g Mode - See 3.8.1.

3.11.2 Accept and Insert Earth-Computed Trajectory Data - Complete backup. See 3.6.5.

3.11.3 Return to Zero g Mode - See 3.7.11.

3.11.4 Perform Spacecraft Attitude Control Functions - See 3.7.1.1.



FOLDOUT FRAME 1

Figure A-5. Mission Success (5)

FOLDOUT FRAME 2

3.11.5 Determine Trajectory Change and ΔV Required - See 3.6.1.

3.11.6 Apply ΔV Correction (as required) - See 3.7.3.

3.11.7 Continue Earth Approach Coast - Passage of time only. Specific functions needed during this time period.

3.11.8 Terminate Experiment Program - Many mission success operations.

3.11.9 Checkout and Maintain All Entry Critical Functions - Each function should be delineated.

Functions Applying to More than One Phase

4.X.X. See 3.6.

3.6.4 See 3.6.

3.6.10 See 3.6.

3.6.11 See 3.6.

Series Functions	
Mission Success	Crew Safety
3.11.3 Return to Zero g Mode	3.11.3 See 3.7.11.
3.11.4 Perform Attitude Control Functions	3.11.4 See 3.7.1.1.
3.11.6 Apply ΔV Correction	3.11.6 See 3.7.3.
3.11.7 Continue Earth Approach Coast	3.11.7 This phase must be accomplished safely, but specific tasks need to be determined.
3.11.8 Terminate Experiment Programs	
3.6.11 Provide Environment and Life Support	3.6.11 See 3.6.

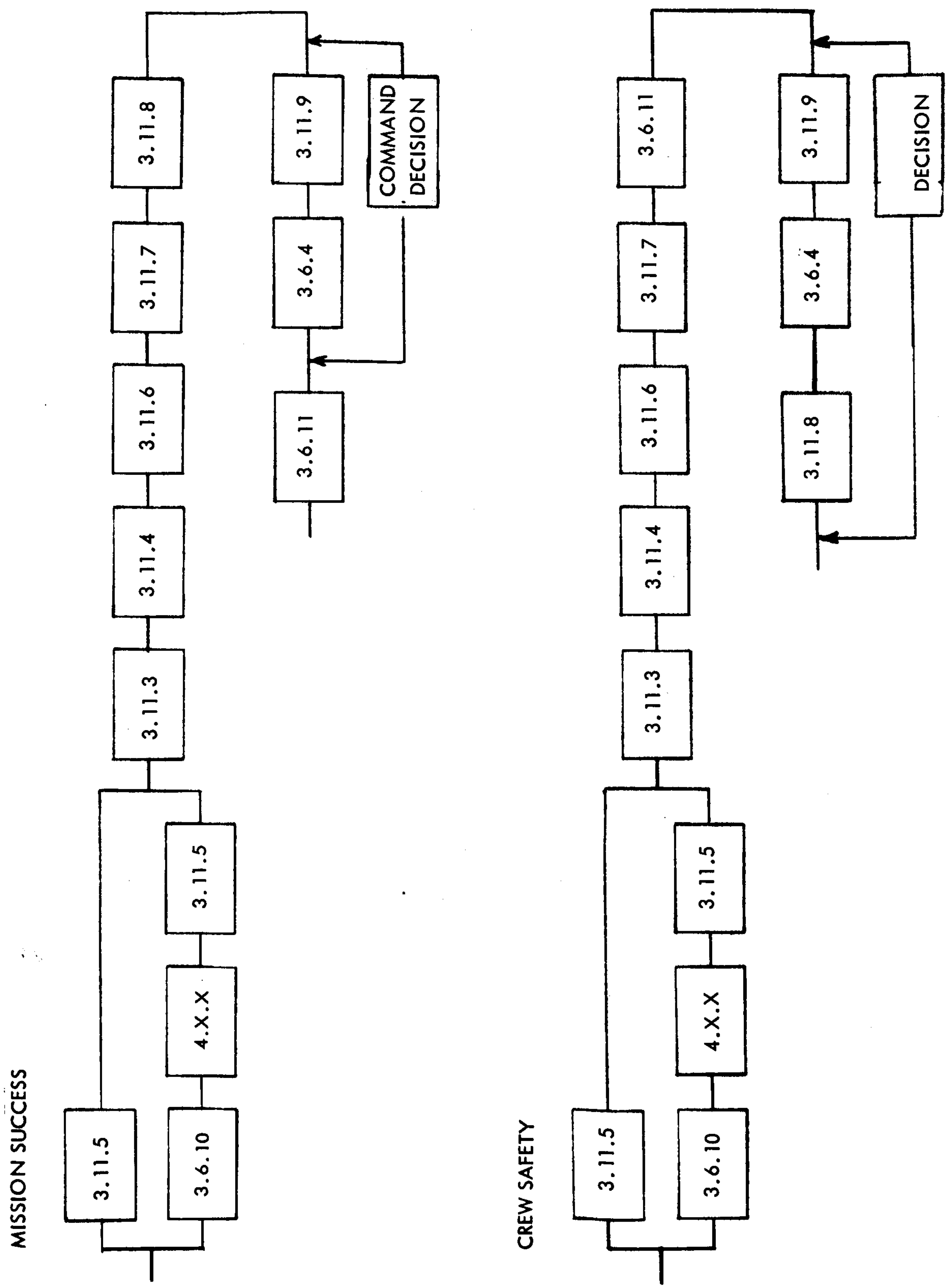


Figure A-6. Mission Success (6)

PHASE 3.12 PERFORM EARTH RETRO-OPERATIONS

3.12.1 Determine Spacecraft Position and Velocity - Redundant with ground based equipment.

3.12.2 Accept and Insert Earth Computed Data - Backup data.

3.12.3 Orient Spacecraft Along Earth Entry Vector - Critical maneuvers for accurate alignment. Using the last propellant to slow the spacecraft as much as possible is desirable, but applied in the wrong direction would send the spacecraft out of the reentry corridor.

3.12.4 Apply Retro-Power to Burn-Out - This function increases the margin for safe reentry by decreasing the spacecraft velocity, but it is not necessary to either mission success or crew safety.

3.12.5 Reorient Spacecraft for Jettison Operation - A wide latitude exists for orientation accuracy for jettison; in fact, jettison may not be necessary.

3.12.6 Jettison Retro-Vehicle. For clean reentry and designed use of the heat shield, jettison of the retro-vehicle is desirable; however, the specific design will dictate the exact method; e.g., (1) for a retro-vehicle which can burn up and fall away from the spacecraft, retention of the retro vehicle would increase the margin on the spacecraft heat shield, and (2) for a spacecraft/retro-vehicle orientation which would allow aerodynamic pull off (higher retro-vehicle drag), the jettison operation is unnecessary. Here, assume design cleverness so that jettison is not required.

3.12.7 Continue Coast to Earth Entry - Time only.

4.X.X. See 3.6.

3.6.4 See 3.6.

3.6.10 See 3.6.

3.6.11 See 3.6.

3.7.1.1 Perform Stability and Control Functions - See 3.7.

Series Functions	
Mission Success	Crew Safety
3.12.7 Continue Coast to Earth Entry	3.12.7 Time only.
Functions Applying to More Than One Stage	
3.7.1.1 Perform Stability and Control Function	3.7.1.1 See 3.7.
3.6.11 Provide Environmental and Life Support Functions	3.6.11

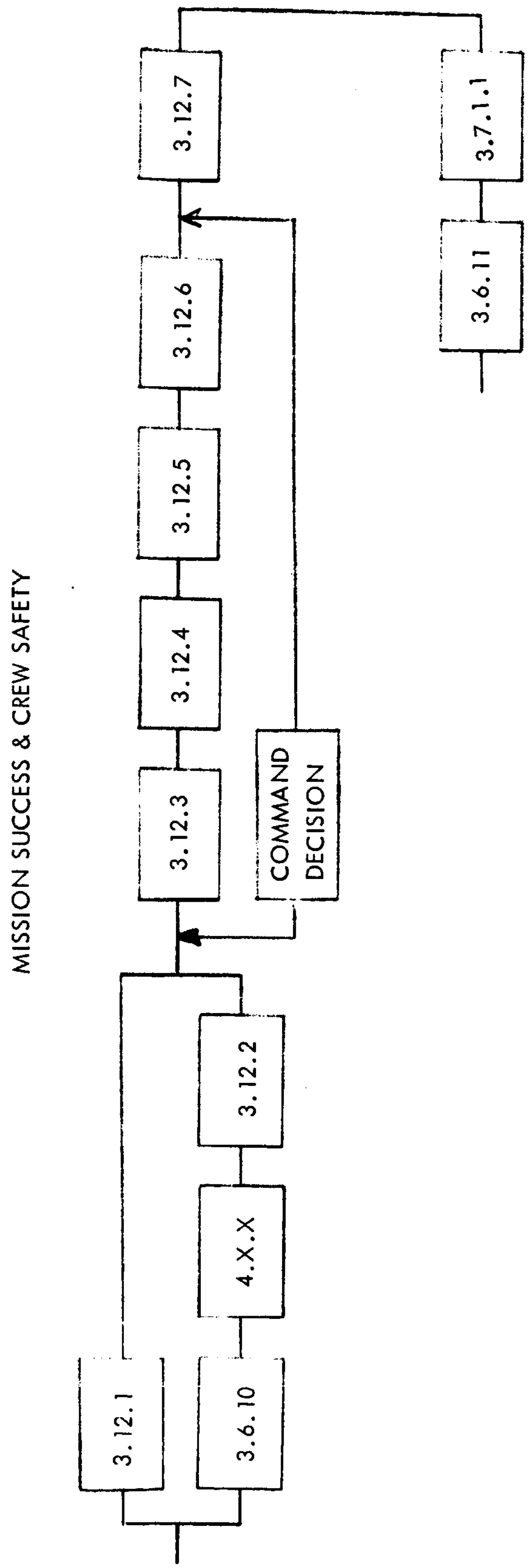


Figure A-7. Mission Success (7)